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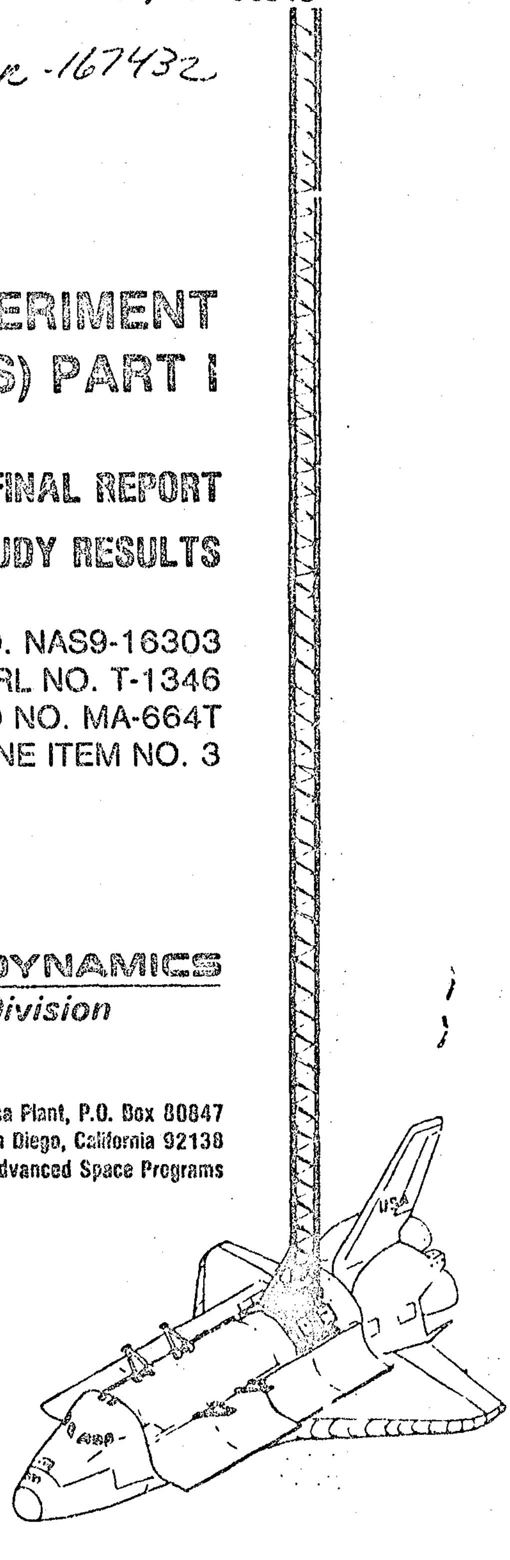
SPACE CONSTRUCTION EXPERIMENT DEFINITION STUDY (SCEDS) PART I

FINAL REPORT

VOLUME II • STUDY RESULTS

CONTRACT NO. NAS9-16303
DRL NO. T-1346
DRD NO. MA-664T
LINE ITEM NO. 3



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Advanced Space Programs

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SPACE CONSTRUCTION EXPERIMENT DEFINITION STUDY (SCEDS) PART I

FINAL REPORT

VOLUME II • STUDY RESULTS

1 September 1981

Submitted to
National Aeronautics and Space Administration
LYNDON B. JOHNSON SPACE CENTER
Houston, Texas 77058

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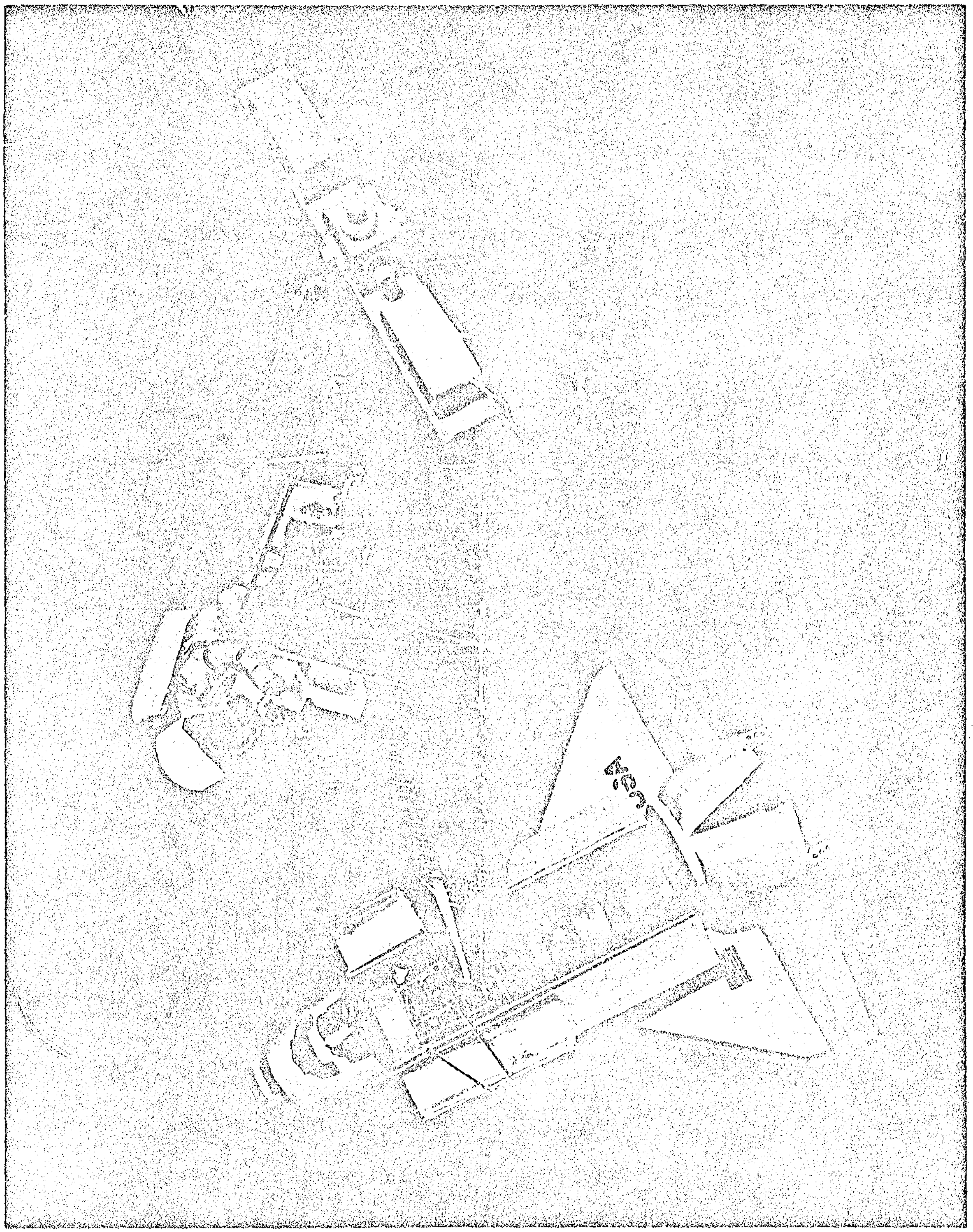

FOREWORD

The final report was prepared by General Dynamics Convair Division for NASA-JSC in accordance with Contract NAS9-16303, DRL No. T-1346, DRD No. MA-664T, Line Item No. 3. It consists of two volumes: (I) a brief Executive Summary and (II) a comprehensive set of Study Results.

The principal study results were developed from February 1981 through July 1981 followed by final documentation. Reviews were presented at JSC on 1 May 1981 and 21 July 1981.

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TABLE OF CONTENTS

TABLE OF CONTENTS, Contd

TABLE OF CONTENTS, Contd

TABLE OF CONTENTS, Contd

LIST OF FIGURES

LIST OF FIGURES, Contd

LIST OF FIGURES, Contd

LIST OF TABLES

ACRONYMS AND ABBREVIATIONS

A/D	Analog-to-digital
BIU	Bus interface unit
C&W	Caution and warning
CCTV	Close circuit television
CDR	Critical design review
CER	Cost estimating relationship
CIS	Centaur-in-shuttle
CRT	Cathode-ray tube
CSDL	Charles Stark Draper Laboratory
D&C	Display and control
DAP	Digital autopilot
DIO	Discrete input-output
DIS	Digital integrating system
DOF	Degrees of freedom
DRD	Data requirements document
DRL	Data requirements list
EVA	Extravehicular activity
FOV	Field of view
FSE	Flight support equipment
FSS	Flight support system
GEO	Geostationary earth orbit
GPC	General purpose computer
GSE	Ground support equipment
HEXFET	Trade name for MOS-FET switch (metal oxide semiconductor - field effect transistor)
JSC	Johnson Space Center
KSC	Kennedy Space Center
LEO	Low earth orbit
LSS	Large space system

MDF	Manipulator development facility
MDM	Multiplexer - demultiplexer
MMU	Manned maneuvering unit
MPS	Material processing science
MUX	Multiplexer
NASA	National Aeronautics and Space Administration
OSS	Office of Space Science
OTV	Orbital transfer vehicle
PCM	Pulse code multiplexer
PDI	Payload data interleaver
PDR	Preliminary design review
PIDA	Payload installation and deployment aid
PMP	Parts, materials and processes
PRCS	Primary reaction control system
PROM	Programmable read only memory
PRR	Preliminary requirements review
RAM	Random access memory
RCS	Reaction control system
RMS	Remote manipulator system
ROM	Read only memory
S/C	Spacecraft
SASP	Science and applications space platform
SBR	Space based radar
SCAFEDS	Space construction automated fabrication experiment definition study
SCE	Space construction experiment
SCEDS	Space construction experiment definition study
SIMFAC	Simulation facility
SIO	Serial input-output
SOC	Space operations center

ACRONYMS AND ABBREVIATIONS, Contd

SSLCC	Space system life cycle cost
SSP	Standard switch panel
TSS	Truss support structure
VRCS	Vernier reaction control system
WBS	Work breakdown structure

SECTION 1
INTRODUCTION

1.1 SCOPE

This is the second of two volumes comprising the SCEDS Final Report. It contains the detailed results of all Part I study tasks. Volume I provides an Executive Summary of the study results. This report is the final deliverable contract data item. It satisfies the requirement for Line Item 3 (DRD MA-664T) of DRL T-1346.

This section provides an overview and top level summary of the study objectives, approach, and results.

1.2 STUDY OVERVIEW

The top level objectives of the SCEDS program are:

a. To define a basic Shuttle flight experiment which will provide needed data on construction of large space systems from the Orbiter, where it is not practicable to obtain the data from ground tests. This includes experiments in these areas:

   1. Predicted dynamic behavior of a representative large structure.

   2. On-orbit construction operations.

   3. Orbiter control during and after construction.

b. To identify and define evolutionary or supplemental flight experiments for development or augmentation of a basic flight experiment.

The study activities were divided into six major tasks with appropriate sub-tasks within the major task headings as shown in Figure 1-1. The first half of the study was almost entirely Task 1 wherein candidates for deployable structures, the basic experiment, EVA/RMS operations, and suitcase experiments are identified, described, and evaluated. A study and evaluation of damping techniques for the structure was conducted and a damping augmentation approach was selected. The effects of restowage and return of the experiment were also identified. Task 1 resulted in the selection and recommendation of experiments and concepts that were reviewed for final selection by the joint NASA/JSC, Draper Lab, Convair working team at this midterm review.

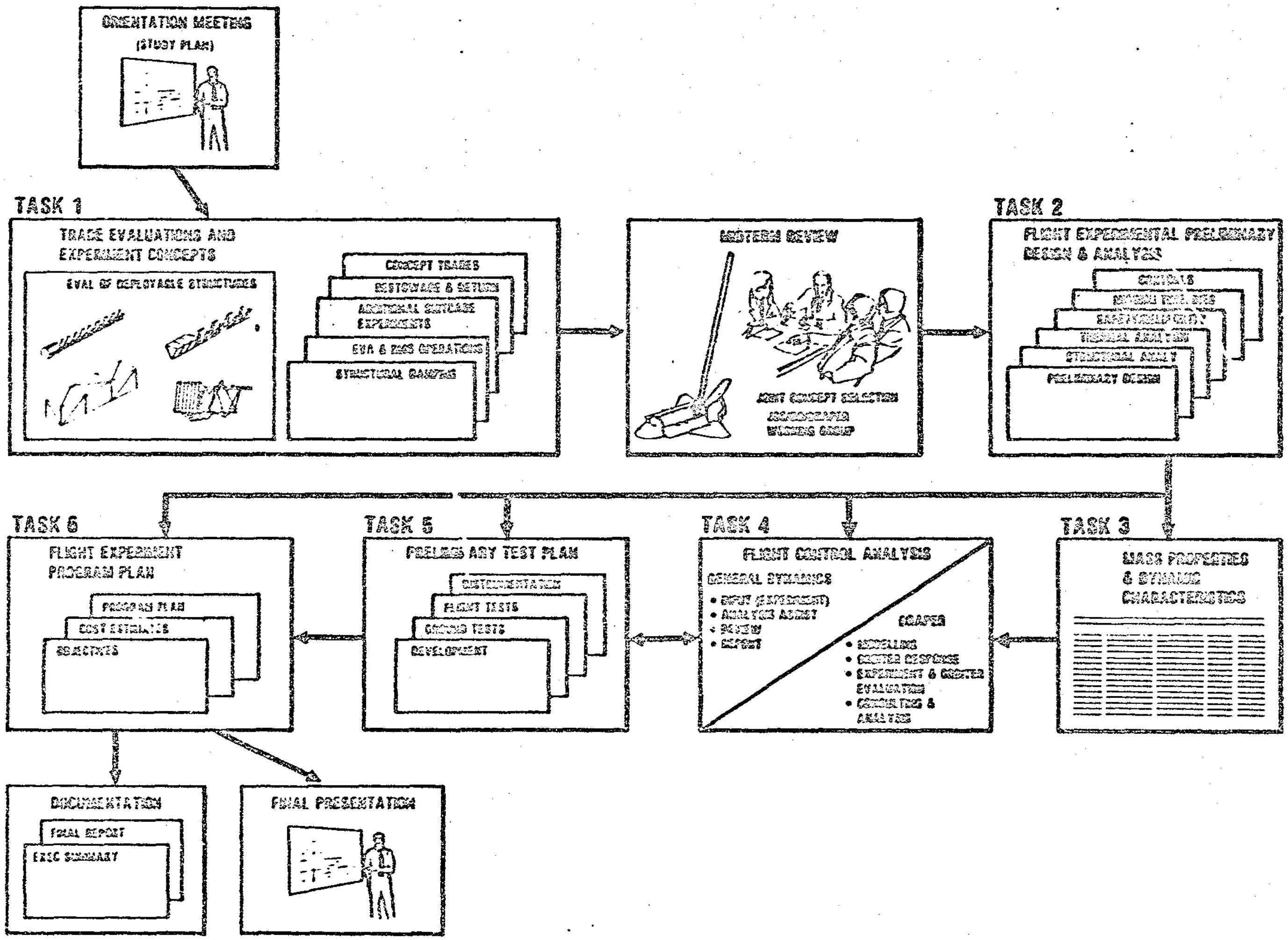

Figure 1-1. SCEDS Part I Study Approach

The selected concepts, tests, experiments, and operations were then used to prepare a preliminary design and analysis in Task 2. The preliminary design data were used to derive mass properties and dynamic modes for the experiment for further analysis by Draper Lab in Task 4. Task 5 generated a preliminary test plan for the Space Construction Experiment, and a program plan and cost estimate for the program were prepared in Task 6.

1.3 SUMMARY

The preliminary design for a basic Space Construction Experiment and concepts for additional suitcase experiments for Extra-Vehicular Activity (EVA) and Remote Manipulator System (RMS) construction operation were developed to incorporate the following characteristics:

a. SCE will share a Shuttle mission with other payloads as a payload of opportunity.

b. SCE will remain attached to the orbiter throughout test. Jettison capability is provided; however, the experiment will normally be restowed and returned to earth by the orbiter.

c. SCE will exercise a variety of appropriate Large Space System (LSS) construction and assembly operations utilizing basic Space Transportation System (STS) capabilities (EVA, RMS, CCTV, ILLUMINATION, etc.) to be correlated with ground tests and simulations.

d. SCE uses representative LSS elements. The basic experiment employs a deployable low natural frequency structure. The structure will have a very low coefficient of thermal expansion achievable through the use of graphite composite materials of construction. Structural dynamic tests will provide data to be correlated with math and ground test models.

e. SCE provides options to approach proven capabilities of the orbiter conservatively and safely exceed proven limits to establish usable capabilities for control, mission timelines, and critical interfaces. These options include variablility of mass moment of inertia and variable damping augmentation.

Details supporting the SCEDS system concept development are presented in the body of this volume. The text is arranged by topic and discipline rather than by individual study task to avoid fragmenting the treatment of specific system definition efforts.

SECTION 2

STRUCTURE

Analysis and trades were performed to define and select a representative deployable Large Space System structure for the basic experiment. A comprehensive list of requirements applicable to near term Large Space Systems candidates was derived and used for evaluation. Candidate structures were selected from known concepts in various stages of development. Two alternate structures were selected for further evaluation as part of the SCE trades and analysis presented in Section 4.

2.1 DEPLOYABLE STRUCTURES REQUIREMENTS

Large Space Systems such as the Space Operations Center (SOC), Geostationary Platform (Geoplatform), Science and Applications Space Platform (SASP), and Space Based Radar (SBR) are being studied and defined today for potential implementation in the near future. In defining requirements for the Space Construction Experiment, these then are the primary applications to consider, if indeed SCE is to be a cornerstone of near term space construction efforts.

Concepts for the aforementioned systems are shown in Figure 2-1. Each represents an integrated modular construction approach, whereby basic system elements such as reflectors, feed modules, habitability modules, and power modules are interconnected through a primary structural element, usually depicted as a deployable truss. It is conceivable that a single deployable truss element could be developed to meet most of the needs of these and other future space platforms. Use of such an element in the SCE will both assure an applicable data base for LSS design and bring the technology for LSS structures to a high initial state of readiness.

The most significant factor in achieving acceptable structural performance is selection of a structural concept with the best overall capabilities and subsequent optimization of its configuration and sizing. A review of selected LSS concepts revealed requirements which were used to evaluate space truss candidates. Table 2-1 summarizes these requirements and indicates which have major importance to the systems considered. The major factors are discussed below:

a. Physical Characteristics. It is generally desirable, and frequently critical, to minimize structural distortions that result from elastic strains and/or thermal expansion/contraction. In large structures such distortion effects can result in high, transient, localized loading of individual structural elements. It can also degrade system performance. These effects can be reduced by selecting structural material having low coefficient of thermal expansion (CTE), low density, and high elastic modulas. Equally effective in reducing deviations from nominal structural shape is selection of a

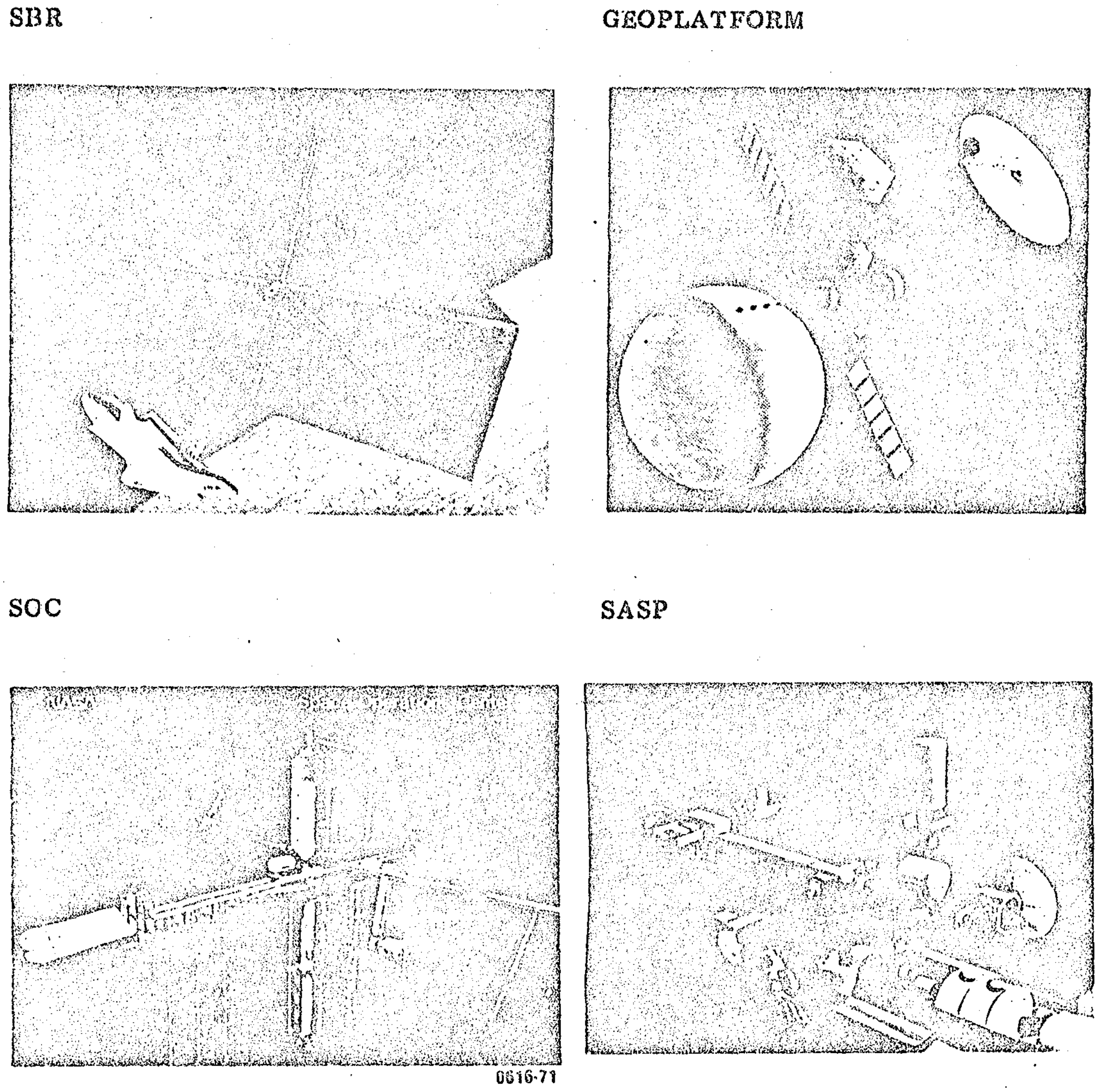

Figure 2-1. Large Space System Candidates for Near Term Applications

Table 2-1. Primary Requirements for Space Truss Concept Evaluation

Deployable Space Structure Requirements	Application: Space operations center	Geostationary platform	Science applications platform	Space radar system	Space construction experiment
Physical characteristics					
• Strength		✓		✓	✓
• Stiffness	✓	✓	✓	✓	✓
• Column stability		✓		✓	
• Thermal stability		✓	✓	✓	
• Structural efficiency (lightweight)		✓		✓	
• Orbital life expectancy		✓		✓	
Stowage & deployment factors					
• Packagability		✓		✓	✓
• Controlled deployment		✓	✓	✓	✓
• Retraction capability				✓	✓
System compatibility factors					
• Suitability as modules for space assembly	✓	✓	✓		✓
• Suitable for hard mounting of substructures & equipment modules	✓	✓	✓	✓	✓
• Compatible with preinstalled hardware & service lines	✓	✓	✓	✓	✓
• Manned traverse capability	✓		✓		✓
Other factors					
• Cost effectiveness					✓
• Reliability (mech & struc)	✓	✓	✓	✓	✓
• Hardware development status					✓
• Applicability to LSS					✓

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structural type and configuration that is inherently and characteristically stable in shape, and that is proportioned and sized to adequately exploit those characteristics.

b. Stowage and Deployment Factors. The structure must be capable of being efficiently packaged for stowage in the launch vehicle, and of being mechanically deployed in a controlled fashion. The deployment sequence must be arrestable and reversible at any stage of deployment and must exhibit a high degree of mechanical reliability, without involvement of Extra Vehicular Activity (EVA). Ability of the structure to withstand shear and bending loads during deployment is essential for Orbiter-attached operations.

c. System Compatibility Factors. In its deployed configuration, and during deployment, the structure must be capable of accommodating and supporting functional subsystems (e.g., preinstalled hardware and service lines).

For applications such as the SOC, where the system size exceeds the size potential of a single deployed structure, it will typically be necessary to individually deploy several such structural "modules" and then integrate them to form a larger assembly. The structure must, therefore, possess suitable interface features which will facilitate such integration.

Similarly, hardpoints on the structure will be needed for the installation of operational subsystems and equipment (e.g., feed, reflector, and array installations).

The structural configuration should also be suitable for accommodation of EVA excursions along the structural modules for purposes of system assembly, inspection, troubleshooting, repair, and maintenance. This implies that the structure must be durable enough to withstand reasonable wear and tear and contact with personnel performing system construction and maintenance operations.

2.2 DEPLOYABLE STRUCTURES SELECTION

A review of available data on deployable structures and LSS technology plus Convair's in-house activities in the design and development of space structures led to selection of ten representative LSS structural beam elements as candidates to be evaluated for applicability to the Space Construction Experiment. These candidates are shown in Figure 2-2.

2.2.1 EVALUATION APPROACH. To establish a fair comparative basis for the structural candidates to be evaluated, it was decided to apply the following conditions to each candidate.

a. Each beam candidate was sized to occupy 90% of the usable Orbiter payload envelope when fully folded/retracted for transportation.

b. The rigid structural elements of each beam candidate were assumed to have a length-to-radius-of-gyration ratio (L/ρ) = 250, probably the maximum value that would be used for LSS applications.

c. All structures were assumed to be made of graphite/epoxy composite material.

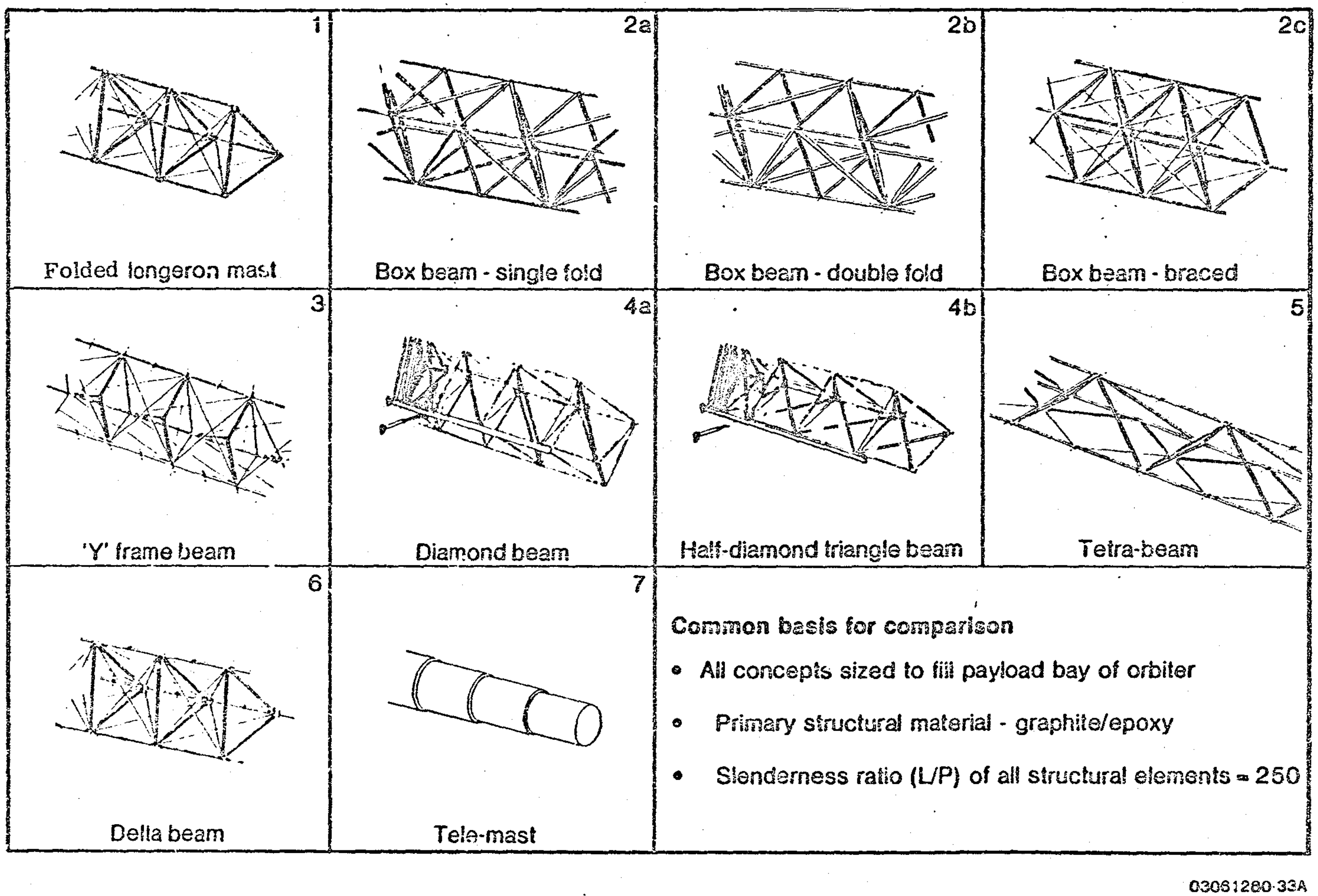


Figure 2-2. Candidate Deployable Structure Concepts

2.2.2 CANDIDATE STRUCTURES DESCRIPTIONS. The selected structures concepts are described as follows:

a. Concept 1 - Folded Astromast with Articulated Longerons (Reference 14). The Articulated Astromast, sized to fully utilize the orbiter payload bay, is shown stowed in the Orbiter in Figure 2-3. Allowing space within the envelope for a surrounding canister, the 3.5m equilateral triangular frames represent the largest size stowable. The mast structure consists, also, of longeron segments pivotally connected to the apices of the triangular frames, and tension ties that cross-brace the three rectangular faces of each bay. The transition from stowed configuration to deployed configuration requires a powered mechanism to extend each bay and activate diagonal cable tensioning latches. This deployment and retraction control of the structure is provided within the canister assembly. If each bay of the structure occupies a 7.6 cm increment of the 16.5 m usable payload bay length, the total number of bays is 216. The deployed length of each bay of structure is approximately 3.2 m, thus, the total deployed length of the mast is 691.3m.

   The limitations and the less attractive characteristics of Concept 1 are:

   1. Due to the extensive use of diagonal cables, the bending and torsion stiffness of the structure will be lower than comparable structures which use rigid diagonal members.

   2. The necessary pretensioning of the tension ties (in the deployed configuration) results in axial preloading of the longerons, which may reduce column loading capability.

   3. Special provision must be made to ensure control of the (slack) tension ties during deployment to prevent fouling.

b. Concept 2 - Folding Box Beam. Three foldable box beam concepts were selected for evaluation. These are described as follows:

   1. Concept 2a. Single Fold Box Beam (Reference 1)
   The single fold box beam (configured and sized to fully utilize the Orbiter payload bay) is illustrated (stowed in the Orbiter) in Figure 2-4. All longitudinal structural members lie approximately parallel to the axis of the Orbiter. In this arrangement, the maximum achievable length of the component structural element is approximately 8.2m, or just less than half the available payload bay length. The maximum achievable package size is 3m square. If the structural elements are 8.6cm square tubes, then 14 bays of structure can be accommodated within the 3m square envelope. This, when deployed, would provide a beam 115.2m long with a rectangular section 8.2m × 3.0m.

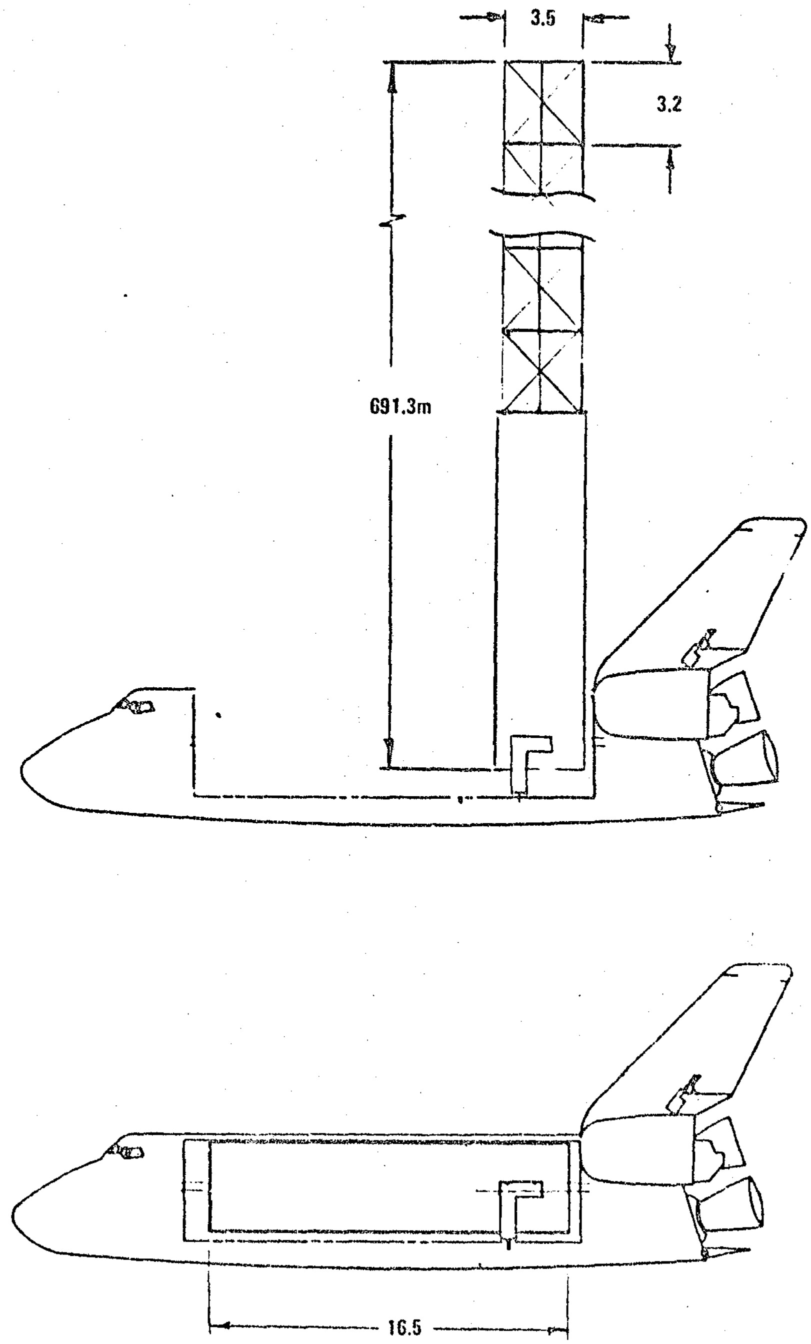


Figure 2-3. Folded Astromast® with Articulated Longerons Maximum Geometry Concept

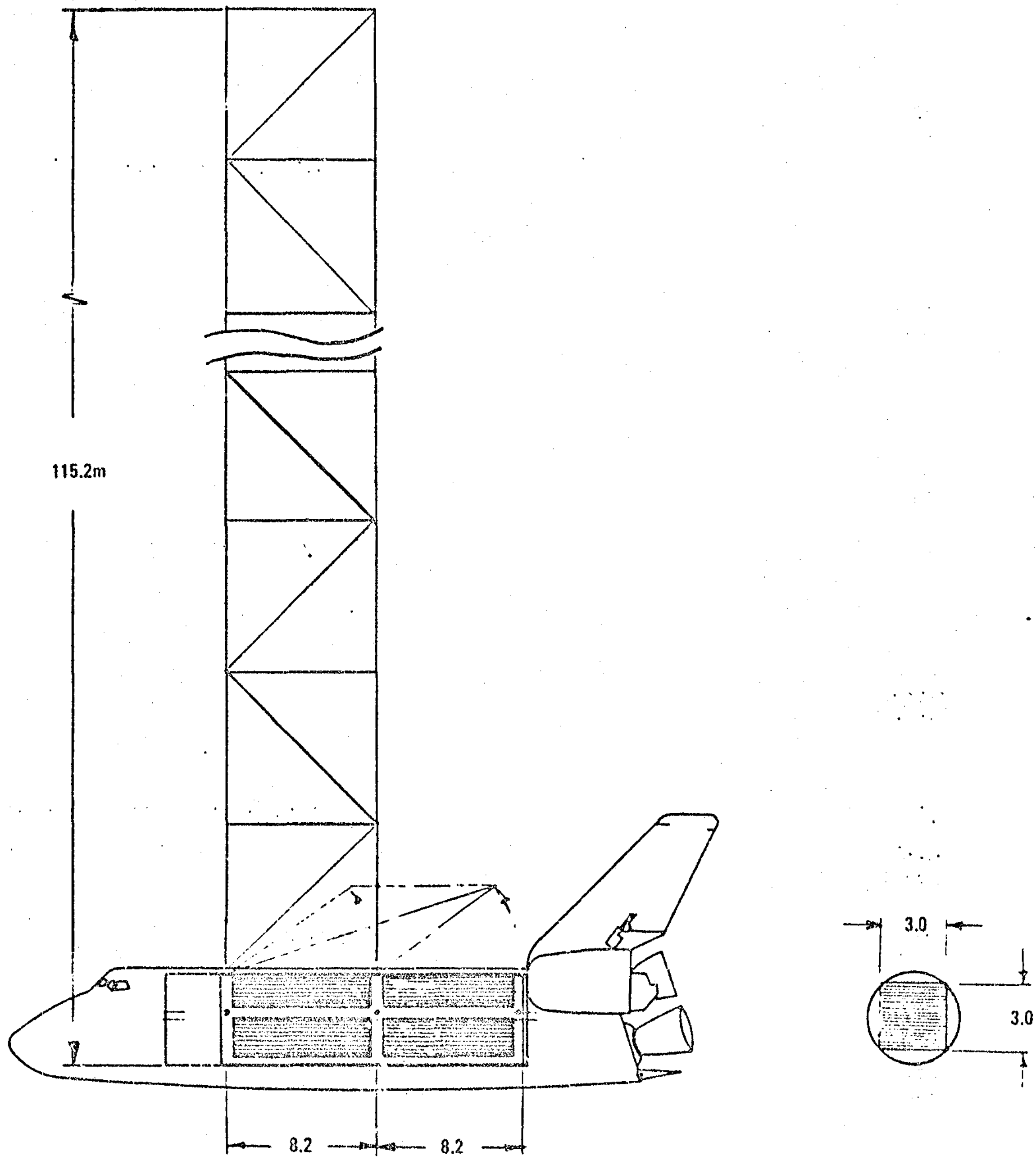


Figure 2-4. Single-fold Box Beam Maximum Geometry Concept

Deployment could be effected incrementally (one bay at a time) using spring-loaded joints or by driving the telescopic diagonal struts, or by means of pull-cable system; however, a complex support structure would be required to react bending loads incurred during deployment. For retraction, the mechanical support equipment would be used to effect reversal of the sequence. To accommodate this equipment, the achievable deployed length (stated above) for the structure would be reduced.

In its deployed configuration, the box beam bay has diagonally braced, rectangular bay facets on all four sides. On two opposite sides, the diagonals are telescopic struts which extend during deployment. With the exception of these telescopic members, the structure has a high efficiency and has conventional structural geometry. Also, packaging density is very low. Even when sized to fully occupy the Orbiter payload bay it weighs only 907 kg.

2. Concept 2b. Double Fold Box Beam (Reference 1)

The double fold version of the box beam concept features telescopic diagonals on all four faces of the structural cells, so that a greater packaging density is achieved by folding in the Y-axis direction, as well as in the Z-axis direction. In this version (shown in Figure 2-5), the maximum achievable length of the component structural element is approximately 5.5m, just less than one-third the usable payload bay length. Tube diameter is 5.72cm for $(L/\rho) = 250$, and thus, 22 bays of structure can be accommodated within the 3.0m square envelope. The deployed size of the structure, therefore, is 120.7m × 120.7m × 5.5m, which is more a platform than a beam.

Since the primary interest is in beams, rather than platforms, a feasible alternative is a version that stows in the form of fourteen separate, stacked, layers rather than as a single, integrated structure. Each layer would then be individually deployed to become a linear 22-cell (bay) beam, 120.7m × 5.5m × 5.5m.

Each beam is deployed, in turn, in two stages. First the entire beam is extended in the Z-axis direction. The second stage is the bay-by-bay unfolding in the Y-axis direction. When all 22 bays are extended, the 5.5m square beam attains its full length of 120.7m. Support equipment required for these events would include a stowage pallet with the capability of elevating the package as the payload is extended; a support frame to interface between the deploying beam and the Orbiter; means of retaining the beams in the vicinity of the Orbiter after deployment and ejection; and means of joining the deployed beams to form the desired structural assembly.

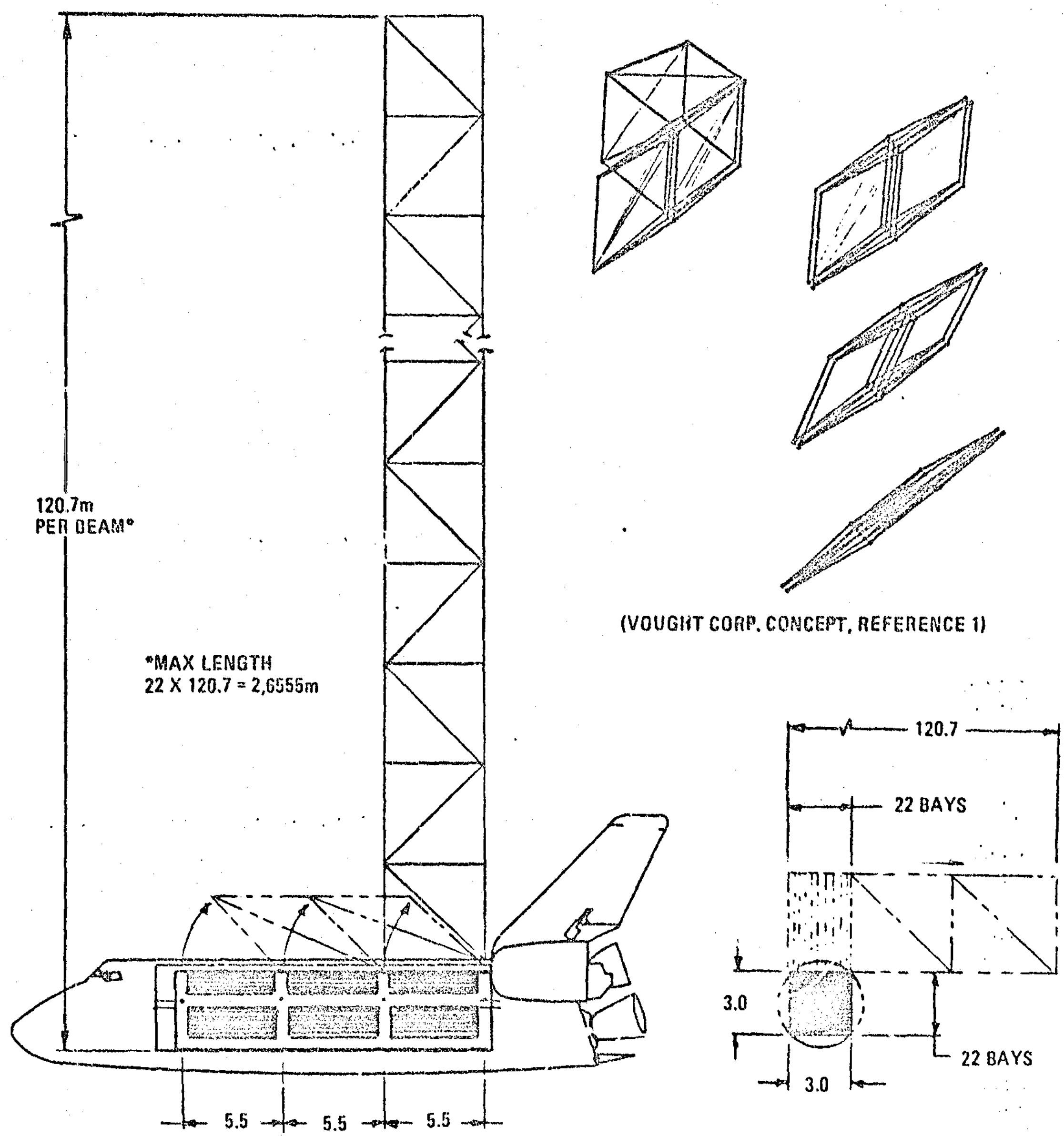


Figure 2-5. Double-fold Box Beam Maximum Geometry Concept

Thus, compared with the single-fold concept, the double-fold version offers significantly greater packaging density, but requires very complex support equipment. The weights of a full payload of the double fold structure would approach the total capability of the Orbiter.

Both the single- and double-fold versions of the box beam require complex pivoted joints at the structural nodes to permit mechanical folding of the structure.

A relatively low reliability rating must result from the extensive use of telescopic struts, which comprise 17% of the structural components in the single-fold version and 34% in the double-fold version.

Fully automated deployment and retraction is feasible, but accommodation of the required support equipment would impact achievable beam size and reliability, due to high complexity of the required deployment mechanisms.

3. <u>Concept 2c. Deployable Box Truss (Reference 22)</u>
The box truss concept, configured and sized to fully utilize the Orbiter payload bay, is illustrated (stowed) in Figure 2-6. All longitudinal structural members lie approximately parallel to the X-axis of the Orbiter. The nominal length of the component structural elements is 8.23m (i.e., just less than half the length of the payload). The cross section size of the stowage envelope is taken to be 3.0m × 3.0m square. If the structural elements are square section tubes with a slenderness ratio of 250, their cross section is 17.17cm square. These stacked tubes represent one bay (cell) of structure; therefore, five bays can be accommodated in the 3.05m width of the envelope, and the number of separate, packaged beams that can be stacked in the 3.05m height of the stowage envelope is four. Thus, such a payload provides four, five bay, square section box beams, each beam having a deployed length of 82.3m and a deployed section of 16.5m × 16.5m. Joined end-to-end, a 329m long beam could be assembled.

For stowage, struts oriented in the X and Z directions have pivotal joints at their midspan point and at their ends. By folding these struts, above these pivots, and simultaneously winding-in the cross-brace tapes, the structure is made to retract. This retraction capability exists in both the X and Z directions, and can be performed simultaneously or in discrete sequence.

For deployment, the first stage is the vertical elevation of the first beam from the payload bay. The second stage is the bay-by-bay extension of the beam in the axial direction. By spring-loading the pivots and by controlling the rate of tape deployment, it may be possible to achieve a degree of controlled self-deployment, but the

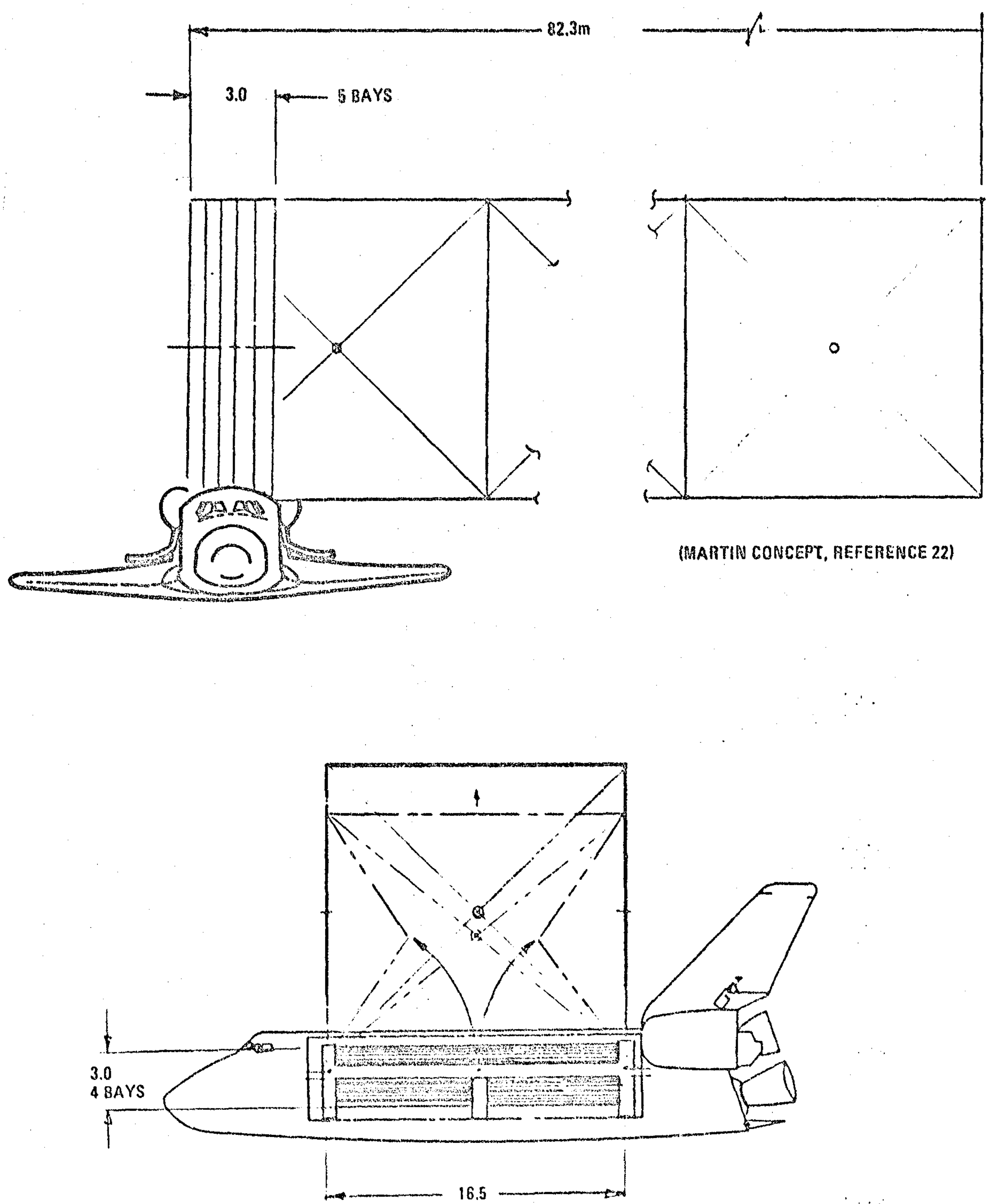


Figure 2-6. Deployable Box Truss Maximum Geometry Concept

bay of structure is basically unstable until its articulated struts have straightened and locked. It would, therefore, be necessary to provide additional support equipment to effect deployment. Provisions must also be made to retain the deployed beams in the vicinity of the Orbiter while the remaining beams are similarly elevated, deployed, and ejected. The weight of such a payload would approach the total capability of the Orbiter.

This structural concept, thus, has efficient packageability and provides a structural geometry of regular cubic cells. Square face cells are not necessarily the most efficient structure, since they are lacking in inherent rigidity. This requires that the square faces must be converted to triangles by the addition of diagonal elements. The approach of triangulating the cross-brace tape cables is appropriate, but represents a significant increase in complexity which tends to decrease mechanical reliability.

Further, since the tapes are relatively long and cannot be substantial in cross sectional area, they still may not provide a sufficient degree of rigidity to the deployed structure. Reliability of tape containment and tension control during the deployment stages is a major concern for this concept.

The provision of cable payout control also provides the potential of controlled retraction of the structure. However, for retraction capability, means must also be provided to unlock and 'break' the strut mid-span pivot joint.

c. Beam Concept 3 - "Y" Frame Mast. This structural concept, shown in Figure 2-7, is basically triangular in section and consists of three longerons, equally spaced frames, and tension ties that provide cross-bracing between frames. To enable folding for packaging, the longeron elements are provided with pivoted joints located midway between frames. Folding is effected by simply rotating the longeron elements about the tips of the frames.

Adjacent longeron elements, thus, rotate in opposite directions to package in accordion fashion. This enables extremely dense, axial packaging with each 2.16m bay of structure reducing to a length increment equal to the 1.8cm diameter of the longeron element.

A beam of this type, designed to fully occupy the Orbiter bay would, therefore, be capable of deploying to a length of $\frac{16.46}{0.018} \times 2.16 = 1975\,m$ and would have a weight of approximately 3175 kg.

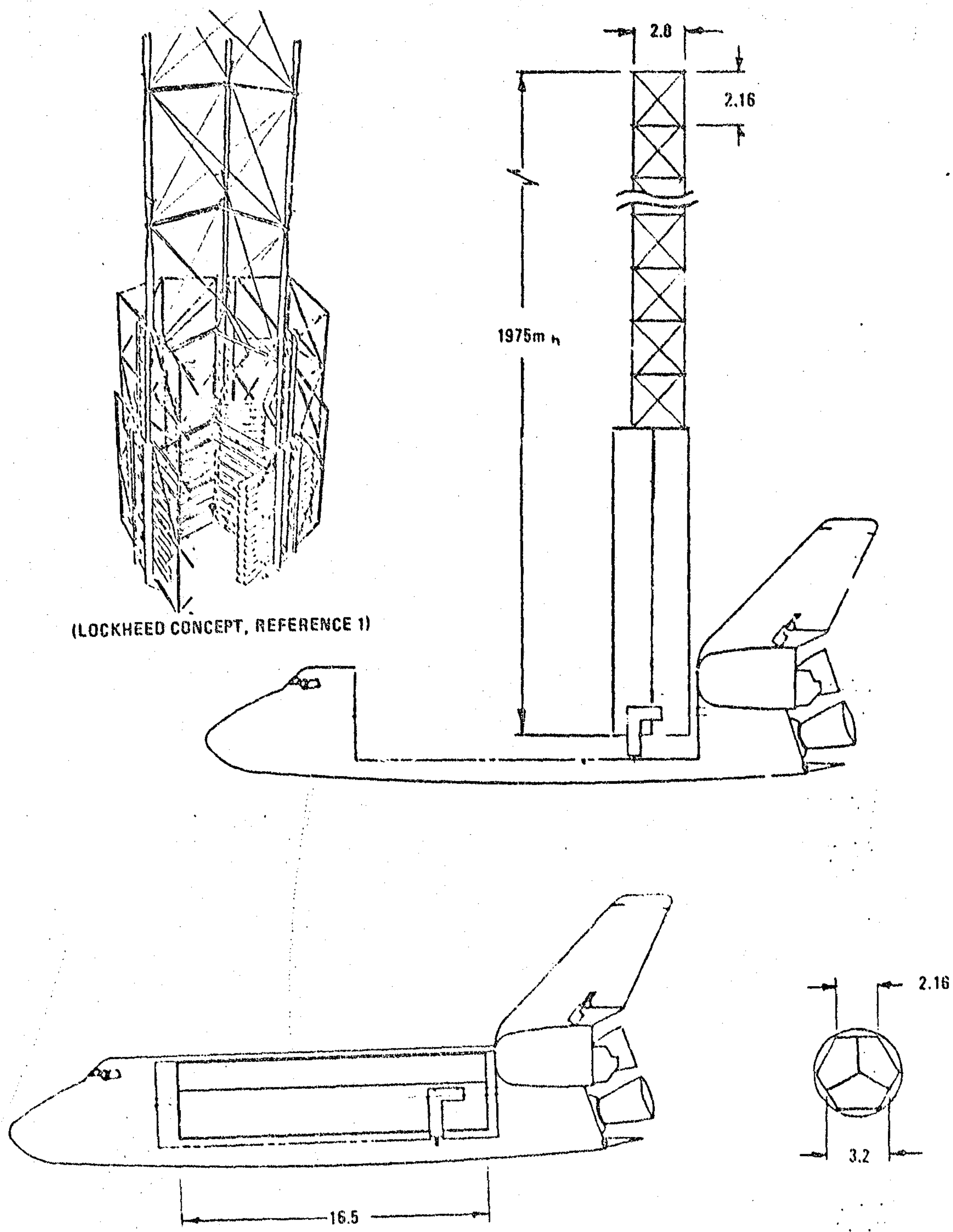


Figure 2-7. Y-Frame Mast Maximum Geometry Concept

The deployment drive mechanism is mounted in the package support cradle which would occupy the full length of the payload bay. The cradle rotates out of the payload bay to stand erect for deployment of the beam. Three guide rails extend from the cradle approximately fifteen feet, to support each successive bay of structure as it transitions through the deployment phase.

The limitations and the less attractive characteristics of the concept are:

1. Due to the extensive use of tension ties, the bending and torsion stiffness of the structure is low compared to similar structures with rigid diagonal members.

2. The necessary pretensioning of the tension ties (in the deployed configuration) results in axial preloading of the longerons, thus reducing column loading capability.

3. Special provision must be made to ensure control of the slack diagonal cables during deployment and retraction to prevent fouling.

4. The deployed cross section size of the beam is limited to approximately one-half of the diameter of the stowage envelope, due to its larger stowed cross section.

d. Diamond and Half-Diamond Concepts

1. Beam Concept 4a. Diamond Beam
This concept features controlled, step-by-step deployment from its initial packaged configuration in which all members lie in parallel orientation. The first stage of deployment is a lateral translation to a diamond-shape. Longitudinal deployment then proceeds, bay by bay. Each bay is driven through its deployment stroke by a pair of actuators in the two side guide rails.

As the mid-jointed members lock straight, the structural bay assumes full rigidity and stability, and is expelled from the guide rails as the following bay goes through a similar deployment sequence. This continues in discrete increments until all bays are deployed and the full-length beam cantilevers rigidly from the guide rails. The deployment sequence can be arrested and even reversed, if necessary, at any stage of beam deployment.

The deployed geometry of the concept fully exploits the benefits of triangulation, which gives the structure a high degree of stiffness and structural efficiency. A degree of structural redundancy exists: in each bay any member may be removed without loss of structural integrity of the remaining structure.

Figure 2-8 illustrates the maximum size potential of the concept. The Orbiter payload bay is shown loaded to maximum (90%) capacity.

The component struts of the packaged beams are oriented transversely in the payload bay within the 3.05m square envelope. The slenderness ratio groundrule sizes the tubes to 2.54cm diameter. Since there are seven tubes in the stack, the height of each package is 0.18m, and seventeen beams can be stacked within the height envelope.

In the longitudinal direction, each structural bay folds to an effective length approximately equal to four tube diameters, or 10.2cm. Therefore, number of bays per beam packageable in the 16.46m payload bay is 161, and the deployed length is 490.7m per beam module. The seventeen beams that comprise the total payload could be joined end-to-end to form a long beam, side-by-side to form a platform, or at angles in T, L, or Δ relationships.

The deployed section shape is diamond (two equilateral triangles back-to-back). A square section shape can be readily achieved if desired by lengthening the internal cross members. Some improvement in beam section properties also results.

2. Beam Concept 4b. Half-Diamond Triangular Truss
This concept is similar to concept 4a except that the section is triangular in shape whereas concept 4a is double-triangular, or 'diamond' in shape.

This represents a 30% reduction in tubular component count. However, there is no longer redundancy in the structure, and foldable cross-braced struts or diagonal tension cables must now be added to stabilize the rectangular face of each bay. The introduction of tension cables into the primary structure system represents some reduction in torsional rigidity, shear rigidity, and thermal stability, but the reduced section packages more compactly, permitting a 76% increase in the number of beams stowable.

Since there are now only four tubes in the stack, the height of each package is only 0.1 meter, so thirty can be stowed in the 3.05m height of the stowage envelope. Thirty such beams joined end-to-end would have an overall length of 14,810 m.

e. Beam Concept 5 - Tetra Beam. This linearly extendable beam is illustrated in Figure 2-9. The two inclined structural faces have triangular geometry while the third consists of cross-braced rectangular panels.

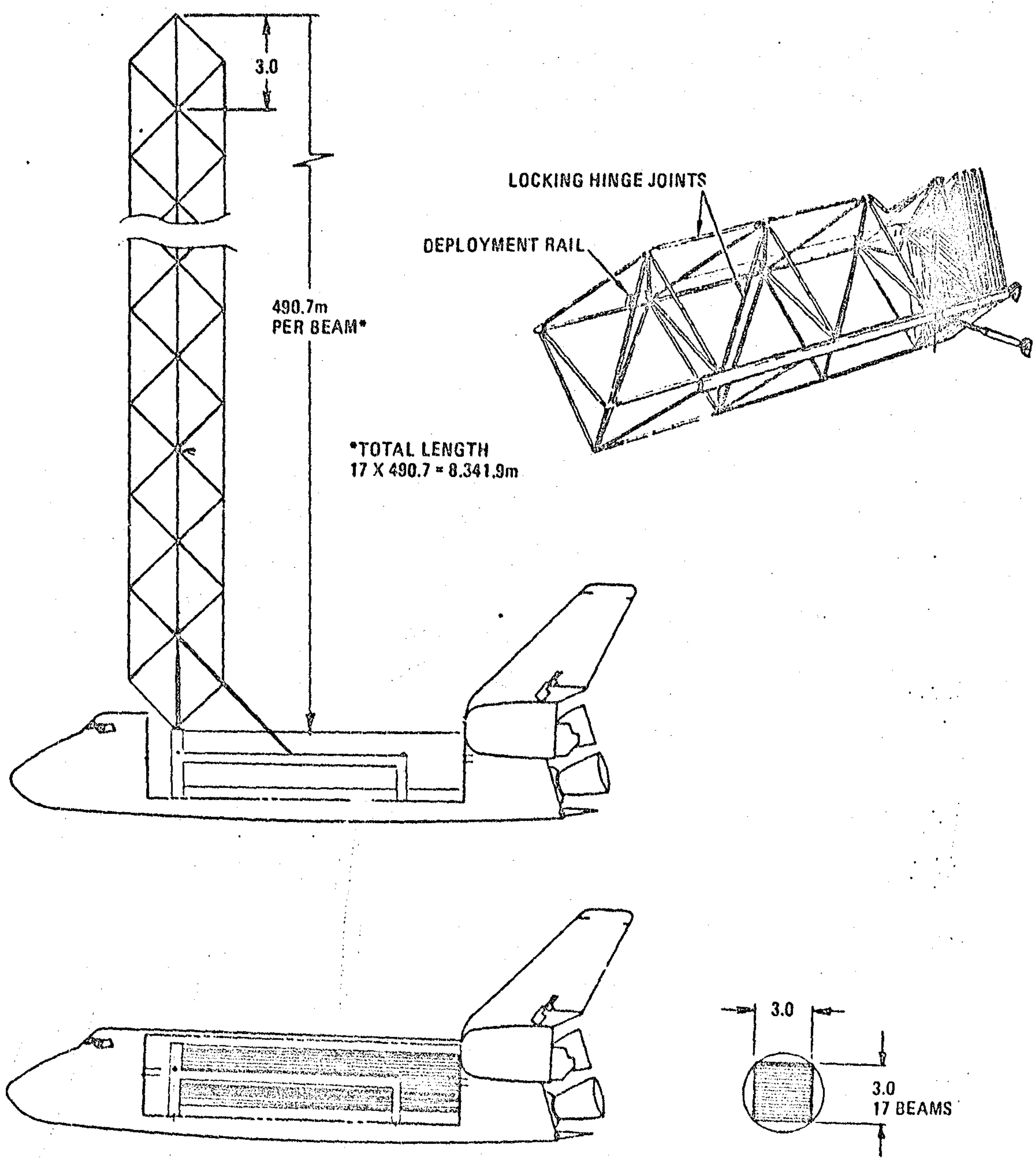


Figure 2-8. Diamond Beam Maximum Geometry Concept

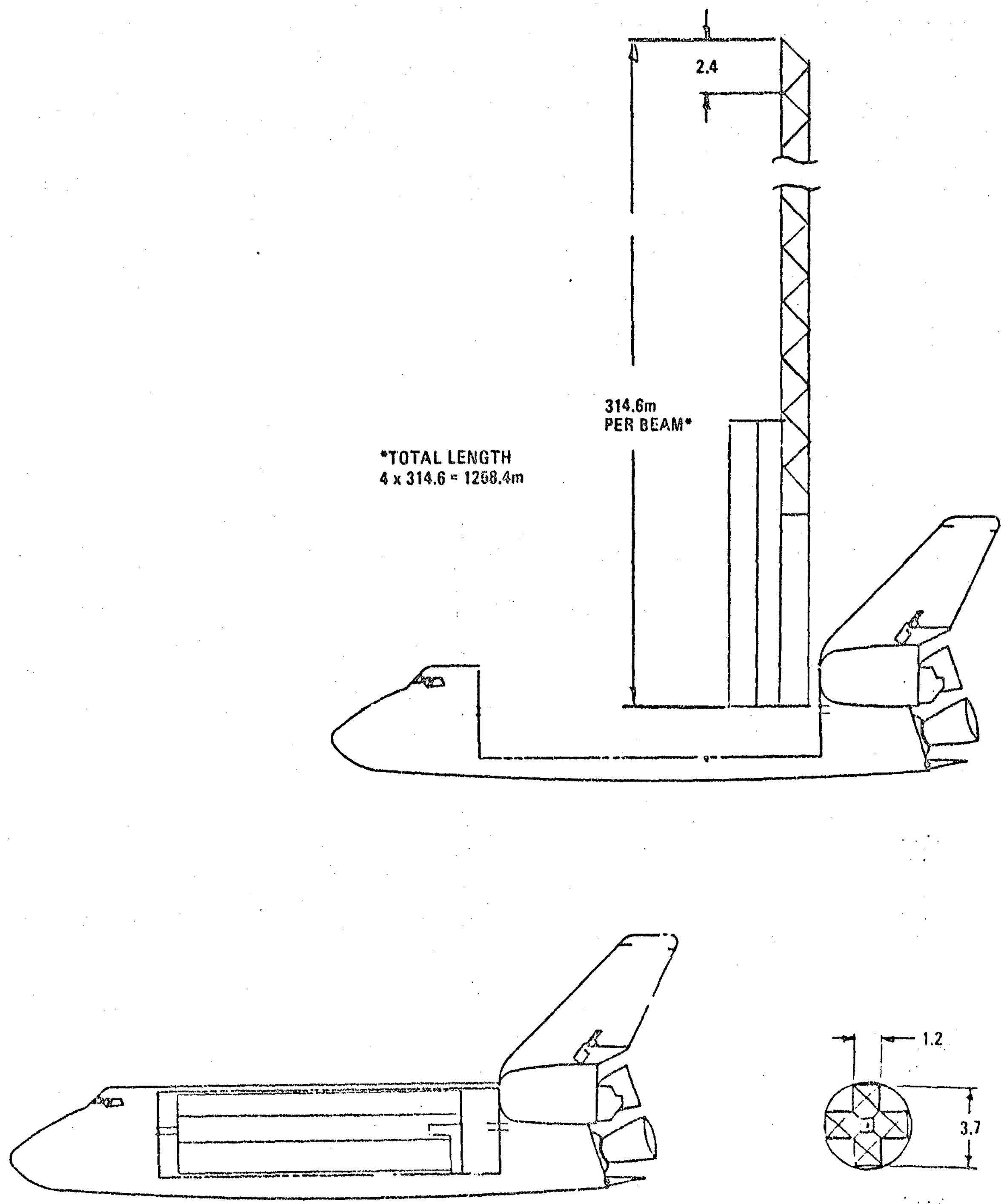


Figure 2-9. Tetra Beam Maximum Geometry Concept

Each longitudinal member has pivot joints at its ends and its mid-span point which enables folding. Figure 2-9 shows a packaging arrangement with four such beams sized to occupy the Shuttle Orbiter payload envelope. In the fully packaged stage, each bay of structure contracts to an effective length of 12.7cm. Assuming an available payload bay length of 16.46m, the maximum beam length stowable would have 129 bays, giving a deployed length of 314.6m.

f. Beam Concept 6 - Delta Beam. This triangular section, shown in Figure 2-10, is characterized by rectangular facets on all three structural faces, stabilized by cross-brace tension cables. Each longeron segment is provided with pivot hinges at the batten frames and at mid-span. These pivots allow the longeron segments to be folded inward, thus drawing the frames together to achieve the packaged configuration. By carefully nesting the folded longeron segment within the frames, a packaged length per bay of two diameters can be achieved, i.e., $2 \times 3.33 = 6.66$cm. The number of bays stowable in the 16.46m payload length is $\frac{16.46 \times 100}{6.66} = 247$. Therefore, total deployed length is $247 \times 4\text{m} = 988\text{m}$.

g. Beam Concept 7 - Tele-Mast. This concept consists of six telescoping segments each 16.46m long. They are graduated in diameter from 4.27m down to 3.35m, so they nest comfortably within each other with a nominal clearance of 7.6cm. Deployment is effected in a telescopic manner by progressively extending the segments by means of a drive assembly consisting of a floating nut on a rotating, threaded drive shaft as shown in Figure 2-11.

Deployment takes place by rotating the screw, which in turn causes the nut to translate. The nut contains a mechanism which unlatches all sections, plus a hook element that captures the base of each section and provides a means to drive the sections outward.

When the first section nears the end of its travel, it latches with the upper flange of the adjacent section. When latching is complete, the drive unit reverses direction, causing the nut element to return down the screw until it unlatches the second section and engages its base. The drive unit again reverses direction and drives the second section outward in the same manner as the first section. The sequence is repeated until all sections are deployed.

Provisions can be made to permit unlatching and return of the mast to its packaged state.

Deployed length is $6 \times 16.46 = 98.8$m. If the nominal wall thickness is 0.025", the total weight of the mast, including the drive system and the latching/unlatching devices would be approximately 4082 kg.

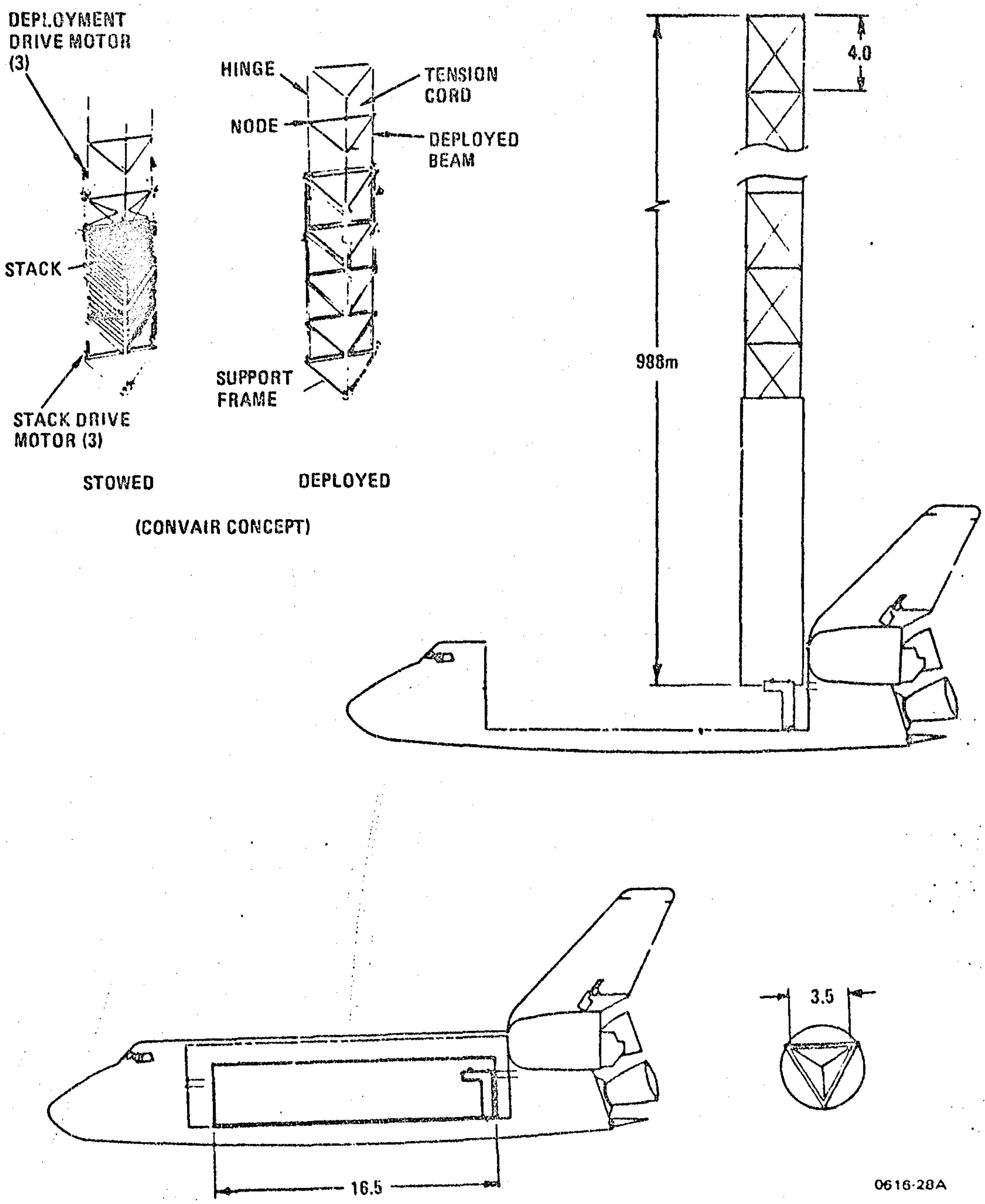


Figure 2-10. Delta Beam Maximum Geometry Concept

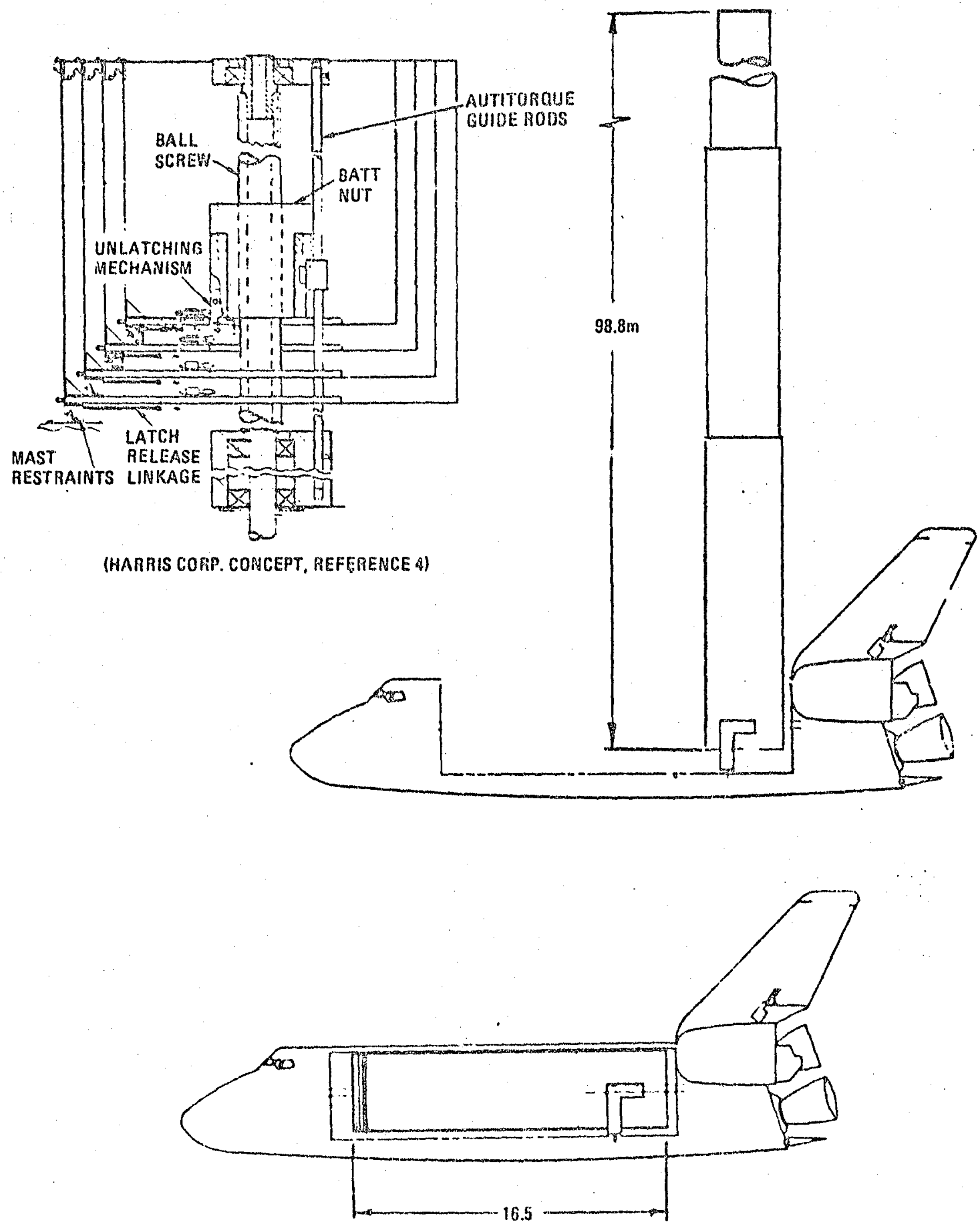


Figure 2-11. Tele-Mast Deployment Mechanism

2.2.3 DEPLOYABLE STRUCTURES EVALUATION. Each of the candidate deployable space structures beams were assessed on a scale of 1 to 10 with respect to the selected evaluation criteria as shown in Table 2-2. In addition, a weighting factor which emphasized the relative importance of each of the criteria was applied.

Table 2-2. Numerical Evaluation of Structures Concepts

Evaluation criteria	Wt factor X	Rating (Y) Scale = 1-10									
		Concept									
		1	2a	2b	2c	3	4a	4b	5	6	7
Packagability (V_D/V_P)	7	5	2	8	4	5	9	10	4	4	2
Strength	4	5	7	7	6	5	7	6	6	5	9
Stiffness	6	4	7	7	4	3	8	7	6	3	10
Column stability	6	2	8	8	5	5	8	7	7	5	10
Thermal stability	6	4	9	10	4	4	10	10	10	5	10
Reliability (mech & struct)	7	5	7	4	5	5	8	6	6	5	7
Controlled deployment	7	9	4	5	6	9	10	10	7	5	9
Retraction capability	4	9	3	3	4	5	9	9	4	4	9
Suitability as modules for space assembly	5	6	8	8	8	7	10	9	7	7	5
Suitable for hard mounting of substructures & equipment modules	5	5	8	8	8	6	9	8	8	8	5
Compatible with preinstalled hardware & service lines	5	5	7	7	7	7	7	7	7	7	4
Manned traverse capability	4	4	5	5	5	5	9	9	6	7	4
Structural efficiency (light wt)	4	5	9	6	7	7	7	8	8	6	5
Cost effectiveness	6	5	2	7	7	6	8	7	3	4	7
Orbital life expectancy	5	4	8	8	4	4	8	4	8	4	8
Hardware development status	3	10	5	5	4	4	8	5	4	5	7
Applicability to LSS	5	4	6	10	9	5	10	10	9	5	4
ΣXY		465	529	615	505	486	763	701	578	458	607

The sums of the products of rating factors and weighting factors resulted in the relative ranking of candidate structures as shown in Table 2-3. This evaluation shows the tetrahedral diamond cross-section beam to have the best overall capabilities, characteristics, and applicability for LSS applications. The half-diamond triangle beam, with the second best overall rating, offers a lower cost alternative, but with reduced reliability and less than optimal physical characteristics. Concept 4a was carried into the flight experiment concept development phase along with the Concept 4b, as they can be used interchangeably.

Table 2-3. Results of Numerical Rating Analysis

Rankings

Evaluation criteria	Concept									
	1	2a	2b	2c	3	4a	4b	5	6	7
Physical characteristics	10	3	4	7	9	2	6	5	8	1
Stowage & deployment	3	10	6	8	4	2	1	7	9	5
LSS compatibility	7	4	4	4	6	1	2	5	3	8
Other factors	7	9	3	4	8	1	2	6	10	5
Overall ranking	9	6	3	7	8	1	2	5	10	4

SECTION 3

CONSTRUCTION OPERATIONS

An analysis was performed to identify and define significant LSS construction issues and operations concerns that needed to be considered for incorporation in the SCE. These issues and concerns were then used to derive Extravehicular Activity (EVA) and Remote Manipulator System (RMS) operations as well as additional suitcase experiment concepts. A concept for restowage and return of the SCE was also developed. This section presents the results of these analyses and evaluations.

3.1 CONSTRUCTION OPERATIONS ANALYSIS

A review of the LSS data base (Section 10) coupled with direct inputs from Convair personnel actively involved in LSS studies and technology development resulted in the identification of significant construction issues and operations. These items (listed in Table 3-1) and subsequent experiment concepts were derived based on the criteria, limits, and constraints described in the following subsections.

3.1.1 TYPES OF LSS CONSTRUCTION. Three types of LSS construction using deployable structures and the current Space Transportation System (STS) are envisioned. These may be described as follows:

a. Fully pre-assembled deployable systems delivered to Low Earth Orbit (LEO) by a single Shuttle flight. This type of platform construction requires either no on-orbit assembly operations or minor equipment installation operations. It is subject to automatic checkout after deployment, with contingency EVA repair operations available. Transfer to higher earth orbits such as Geostationary (GEO) is provided by a preattached boost vehicle. This type of construction is applicable to relatively small platforms and antennas.

b. Partially preassembled deployable system segments delivered to LEO by single or multiple Shuttle flights. This type of platform construction requires on-orbit mating of major deployed platform system segments using automated and/or manual techniques. The orbital transfer vehicle (OTV) may be preattached to a platform segment or attached in LEO prior to boosting the platform to a higher orbit. This type of construction would be applied to large, multi-antenna platforms such as the Geostationary Platforms, or to large aperture antennas.

Table 3-1. Significant Space Construction Issues and Operational Concerns

NO.	DESCRIPTION
1.	PACKAGING, STOWAGE & SUPPORT TECHNIQUES FOR DEPLOYABLE STRUCTURES & SYSTEMS EQUIPMENT IN THE ORBITER CARGO BAY
2.	PRE-DEPLOYMENT PREPARATIONS & OPERATIONS
3.	HANDLING, CONTROL & DISPOSITION OF JIGS, FIXTURES, TRACKS, & ACCESSORIES REQUIRED TO DEPLOY & RETRACT STRUCTURES
4.	CONTROL OF STRUCTURAL DEPLOYMENT & RETRACTION
5.	IN-PROCESS QUALITY VERIFICATION & CONDITION MONITORING
6.	CHECKOUT, MAINTENANCE, REPAIR, CONTINGENCY PROCEDURES & EQUIPMENT
7.	ATTACHMENT/JOINING OF MAJOR STRUCTURAL ELEMENTS & SUBSYSTEM MODULES
8.	INSTALLATION OF SUBSYSTEM EQUIPMENT BEFORE, DURING & AFTER DEPLOYMENT OF STRUCTURE
9.	COMBINED EVA/RMS INSTALLATION & ASSEMBLY CAPABILITIES & TECHNIQUES
10.	APPLICATIONS & EFFECTIVENESS OF SPECIAL RMS END EFFECTORS FOR GRASPING, HOLDING, MANIPULATING & TORQUING
11.	EFFECTIVENESS OF ILLUMINATION/VISIBILITY VISUAL AIDS
12.	SEPARATION & RELEASE OF STRUCTURE FROM ORBITER
13.	RE-ATTACHMENT OR BERTHING OF STRUCTURE TO ORBITER
14.	HANDLING & POSITIONING OF STRUCTURE
15.	RESTOWAGE OF DEPLOYABLE STRUCTURES & EQUIPMENT IN ORBITER CARGO BAY
16.	ORBITER INDUCED DYNAMIC EFFECTS ON STRUCTURE, DEPLOYMENT, CONSTRUCTION EQUIPMENT & OPERATIONS
17.	CORRELATION OF PREDICTED STRUCTURAL DYNAMIC MODES & LOADS WITH MEASURED CHARACTERISTICS
18.	INHERENT STRUCTURAL DAMPING CHARACTERISTICS AND ACTIVE DAMPING TECHNIQUES & EQUIPMENT
19.	STRUCTURAL RATTLE AND BACKLASH EFFECTS
20.	STRUCTURAL THERMAL EFFECTS
21.	STRUCTURAL INERTIA & VIBRATION EFFECTS ON ORBITER CONTROL CAPABILITIES & PERFORMANCE

c. Modular large space system elements delivered to LEO by multiple Shuttle flights. This type of platform construction requires on-orbit assembly of modular system elements such as deployable structural elements, support modules, habitability modules, palletized payload modules, etc. Automated construction equipment and techniques as well as manual EVA operations are applicable to this type of construction. It is an appropriate technique for assembling large platforms for LEO applications such as SOC and SASP.

3.1.2 ORBITER CAPABILITIES, LIMITS, AND CONSTRAINTS. This subsection summarizes relevant limits and constraints of the STS Orbiter capabilities to support space construction. These data were obtained from References 4, 5, 6, 15, 17, and 23. The limits of the Orbiter Digital Autopilot (DAP) are discussed in Subsection 6.3.

3.1.2.1 STS Orbiter Operation

a. Mission

1. Maximum mission duration without special provisions: 7 days.
2. Nominal crew complement: 4 (commander/pilot/mission specialist/payload specialist).
3. Standard accommodations in cabin: 28 man/days (MD) plus 96-hour contingency.
4. Nominal mission orbit: 28.5 to 56 deg inclination/160 n.mi. circular orbit.
5. Launch to orbit time ~45 min (160 n.mi. orbit).
6. Deorbit to landing time ~5 min.

b. Payload Support

1. Visual interfaces:
   - Aft cabin viewing windows
   - CCTV - total of five cameras
   - Six payload bay lights and one RMS light
2. Structural attachments
   - Bridge fittings at sides and at keel (see ICD 2-19001, Reference 5)
3. Electrical interfaces:
   - Power, control, and signal connections are provided at forward and aft payload bay bulkheads and along the sides of the bay (see ICD 2-19001, Reference 5)

4. Fluid interfaces

   - Shuttle provides coolant for thermal control of payloads.

5. Control interfaces

   - Aft flight deck mission and payload specialist stations have provisions for installation of payload unique control and display panels.

3.1.2.2 EVA Operations

a. Time

1. Maximum EVA periods $\leq$ 6 hours
2. One EVA per 24 hour period except for short duration EVA periods where two EVA can be done within 24 hours.
3. Pre- and post-EVA activities require 5 hours.
   - 3.5 hours for EVA prep (includes prebreathing)
   - 1.5 hours for post-EVA operations
4. Recharge of Extravehicular Mobility Unit (EMU) batteries requires 6 hours.
5. Recharge of EMU can be accomplished in 1 hour using new batteries and L104 cartridge.

b. Safety

1. Handrails and/or handholds required for crewman translation.
2. One or two crewmen. If one, second to be on standby.
3. Crewman and equipment/tools to be firmly secured or tethered at all times. Tether attach points required along translation routes and at EVA work sites.
4. All equipment transported or handled to be provided a safety tether attach point.
5. Translation paths to be unobstructed to avoid contact between EMU hardware and vehicle/payload structures.
6. Ensure compatibility of vehicle/payload systems/structure with EVA crewman's EMU. Preclude sharp edges or protrusions and use of hazardous materials.
7. Provide adequate lighting in planning EVA and good visibility in both the "light" and "dark" sides of earth orbit.

c. Physiology

1. Restraints are to be provided at work sites.

   - Foot restraints to EVA functions requiring moderate to heavy force.
   - Handholds and/or tethers for low force tasks.
   - No EVA task to be performed in free-float condition.

2. EMU gloves restrict hand functions and must be considered when gloved hand operations are required.

3. Repetitive manipulations requiring one or two hands and functions requiring controlled body positions should include foot restraints at work site.

4. Equipment decals are required for identifying EVA interface with hardware.

   - EVA work site operations instructions/procedures
   - Equipment identification
   - Contingency procedures
   - Hazardous area/hardware identification

d. Shuttle Support. Orbiter provides equipment and expendables to support EVA, as follows:

1. Two 2-man FVAs of 6 hours duration each, including EMU and expendables.

2. EVA standard hand tools and portable work stations available for use on payload EVA operations.

3. MMU available to payload operations by special request. Space and weight chargeable to payload.

4. Supplemental lighting and TV camera are available to support payload operations.

e. Ground Test Facility. Orbiter-Weightless Environment Training Facility (WETF), a water immersion facility, provides a zero-g environment for training and testing in EVA procedures. Facility includes full scale representation of crew cabin, mid-deck, airlock, and cargo bay door.

3.1.2.3 RMS Operations

a. Shuttle Support

1. One RMS provided as baseline.
2. Second RMS can be requested and is chargeable to payload.
3. Only one RMS can be operated at any time.
4. Two TV cameras provided with RMS, one at elbow and other at wrist.
5. Illumination light provided at wrist joint.
6. RMS control effected by operator from RMS D&C panel in aft flight deck.
7. Standard end effector provided with RMS. Grapple fixture to be supplied by payload.
8. Special adaptive end effectors to be payload supplied.
9. Electric connector provided for control of special end effector.

b. Operational Capabilities

1. RMS is 15.24m (50 ft) long and is normally mounted on the port side cargo bay longeron at Station Xo 1725.9cm along buttock line -Yo 274.3cm and waterline Zo 1129.8cm.
2. Capable of deploying payload weights to 29,483 kg and retrieving 14,515 kg.
3. Maximum velocity of unloaded RMS is 0.61m/sec.
4. Loaded RMS is controlled such that kinetic energy of payload does not exceed a weight of 14,515 kg moving at 0.061m/sec.
5. In automatic mode, RMS can position end effector within ±5.08cm and ±1° relative to shoulder attach point.
6. Zero impulse is applied by end effector at release.

c. Payload Constraints

1. Grapple fixture location within five percent of the payload length of the X-Z plane of payload center of mass.
2. Payload natural frequency at grapple fixture to be no less than 5 Hz (waivers may be applied if simulations show acceptable system response).
3. Payload-to-Orbiter relative velocities to be less than 0.1 deg/sec per axis. Resultant grapple fixture translation to be 3.05 cm/sec maximum relative to shoulder attach point.

4. Reaction control system (RCS) to be deactivated during capture process and maneuvering by RMS.

5. Constrained motion effects are currently under evaluation.

d. Time

1. Preparation of RMS for operation (including power up, uncradle, and checkout) is 60 minutes.

2. Operating in direct and/or reflected sunlight limits operating time of RMS.

   - In cargo bay ~30 minutes.
   - Outside cargo bay ~120 minutes.

e. Ground Test Facility/Simulator

1. Air bearing supported RMS with 3DOF (X and Z-axis and wrist rotation). Shoulder mounting rotated 90° wherein X-Z plane is horizontal. Facility located at manufacturer's plant.

2. An RMS task trainer at JSC. The Manipulator Development Facility (MDF) consists of aft crew station mockup, cargo bay mockup, and a mechanically-operated RMS. User provides helium inflatable payload models. Facility provides environment for training on payload grappling, berthing, visual operations, cargo bay camera operations, and manipulator software operations.

3. A Shuttle mission real time simulation facility (SIMFAC) which provides full fidelity aft crew stations. Includes capability to simulate RMS dynamic operations using computer-generated imagery.

3.1.2.4 Payload Installation and Deployment Aid (PIDA)

a. Operational Capability (conceptual, pre-prototype only)

1. Used as installation and deployment and rotates payload in to/out of cargo bay.

2. Angular travel of 55.5° for boom.

3. Berthing adapter/docking mechanism rotates ~160°.

4. Aids RMS for deployment and installation of payloads from/into Orbiter cargo bay.

5. Can be used to support structures/payload out from cargo bay, freeing RMS to perform other functions.

3.1.2.5 Manned Maneuvering Unit (MMU)

a. Purpose

1. Propulsive backpack using GN_2 thrusters.

2. Gives EVA crewman capability to reach areas outside cargo bay not accessible by other means.

3. MMU and its flight support station stowed in cargo bay adjacent to BHD Xo 1463cm.

4. One or two MMUs can be accommodated/mission-chargeable to payload.

b. Capabilities

1. Attaches to EMU - donned, doffed, and serviced by one EVA crewman.

2. Provides 6DOF control authority and automatic hold capability.

3. Two electrical outlets (2 amp/28 VDC) provided for use on ancillary equipment (e.g., power tools, lights, cameras, sensors).

4. Can carry cargo up to 100 kg size.

5. Total ΔV capability for rotation and translation is 20m/sec.

6. Propellant for recharge and spare batteries can be carried in support station.

7. Nominal operating range is 1000m from Orbiter.

8. Time to don, unstow, and check out MMU is ~15 minutes. Similar time to restow and doff at end of EVA.

9. Propellant recharge time is ~20 minutes.

10. Response: translational accel $9.1 \pm 1.5 cm/sec^2$, and rotational accel $10.0 \pm 3.0 deg/sec^2$.

3.1.3 EXPERIMENTS DESIGN CRITERIA. The following criteria were developed to assist in the identification and definition of candidate space construction tests and operations experiments concepts.

a. Flight or in-space verification experiments shall be performed when ground test is not economically practical or cannot satisfactorily simulate the following:

1. Space environment (zero-g/thermal/vacuum).
2. Orbiter interactions.
3. EVA/RMS operations.

b. EVA shall be an integral part of space construction when it is proven that man-in-the-loop provides:

   1. Significantly lower cost for space construction hardware and systems.

   2. Reduced time to construct/erect space structures.

   3. Ability to perform unique troubleshooting/repair/checkout operations.

   4. Ability to perform operations/functions beyond capability of Orbiter support equipment.

c. Space experiments shall provide results to verify ground tests and simulations where applicable.

d. Safety of the Orbiter and its crew shall not be compromised.

e. EVA tasks shall be planned to meet safety requirements (tethers/handholds/foot restraints/illumination).

f. Use of the RMS for grasping, handling, and manipulating small structural and subsystem elements requires payload unique special end effectors which can be controlled from the RMS Display and Control panel or by manual techniques.

g. Construction operations beyond reach of RMS require EVA with or without MMU.

h. Suitcase experiments are compact and relatively simple tests which:

   1. Require minimal crew training.
   2. Minimize use of Orbiter services/utilities.
   3. Can be performed in less than two hours.
   4. Can be grouped in a single flight package of multiple experiments or flown as separate individual experiments.
   5. Can be flown on a flight space, weight, and time available basis with primary shuttle payloads.

i. Suitcase experiments shall supplement or augment the basic experiment. Suitcase experiments shall serve two major functions:

1. Provide an early evaluation of key construction issues and operations concerns which can impact design and development of deployable space structures and large space systems.

2. Resolve key construction issues not included in the basic experiments.

3.2 IDENTIFICATION AND EVALUATION OF CANDIDATE TESTS AND EXPERIMENTS CONCEPTS

Based on the space construction issues and operations concerns, limits and constraints, and design criteria, potential candidate tests and experiments were identified and evaluated as described in this subsection. Each major issue and operational concern from Table 3-1 is addressed separately. Recommended tests and experiments are then grouped in generic categories, e.g., RMS, EVA, etc.

3.2.1 ISSUE NO. 1 - PACKAGING, STOWAGE, AND SUPPORT. Deployable structures for LSS applications are usually comprised of struts, fittings, and hinges which fold or are collapsed into an efficient envelope size for transportation to space. The structures have preinstalled subsystems hardware such as electrical and fluid conduits, interface mechanisms, and subsystem modules to minimize on-orbit assembly and construction operations. The collapsed structures are supported by devices used to unfold and deploy them. The deployment devices are supported in the Orbiter either on a unique support cradle or on system substructures.

The packaging, stowage, and support of deployable structures and systems equipment in the Orbiter cargo bay is addressed by the design of the basic experiment. Alternative concepts for the basic experiment are presented and evaluated in Subsection 4.2. On the basis of the selected concept, the packaging, stowage, and support details are delineated in the preliminary design of the SCE presented in Subsection 4.2.

Because this issue is fundamental to all Shuttle payloads, the justification for performing a flight test is best described as a demonstration of the ability to package, integrate, flight test, and return large deployable structures and representative LSS equipment. Although the ability of the equipment to survive the launch and space environments can be proven through ground test, the data and experience base that is achievable only though an experimental flight test is of much greater value and use in the design of future LSSs.

3.2.2 ISSUE NO. 2 - PREDEPLOYMENT PREPARATIONS AND OPERATIONS. Preparing the structure for deployment from the Orbiter cargo bay requires a number of steps such as the following:

a. Unlatching or release of primary support points.

b. Rotation of the packaged structure to the proper deployment angle.

c. Unfolding of lateral structural members.

d. Disengagement of secondary supports, holddown mechanisms, and latches.

e. Rotation or extension of axial deployment support rails or tracks.

The use of automated techniques for performing predeployment preparations and operations would require a number of individual drives and controls. This introduces penalties in cost, weight, and complexity.

The use of the RMS to perform the predeployment sequence was evaluated as a means of simplifying the drives and controls for the SCE. A concept for use of the RMS with standard end effector to rotate the truss and deploy or retract the lateral members is illustrated in Figure 3-1. Linear deployment of the truss, however, will require an automatic drive to assure proper sequencing and control during deployment and retraction.

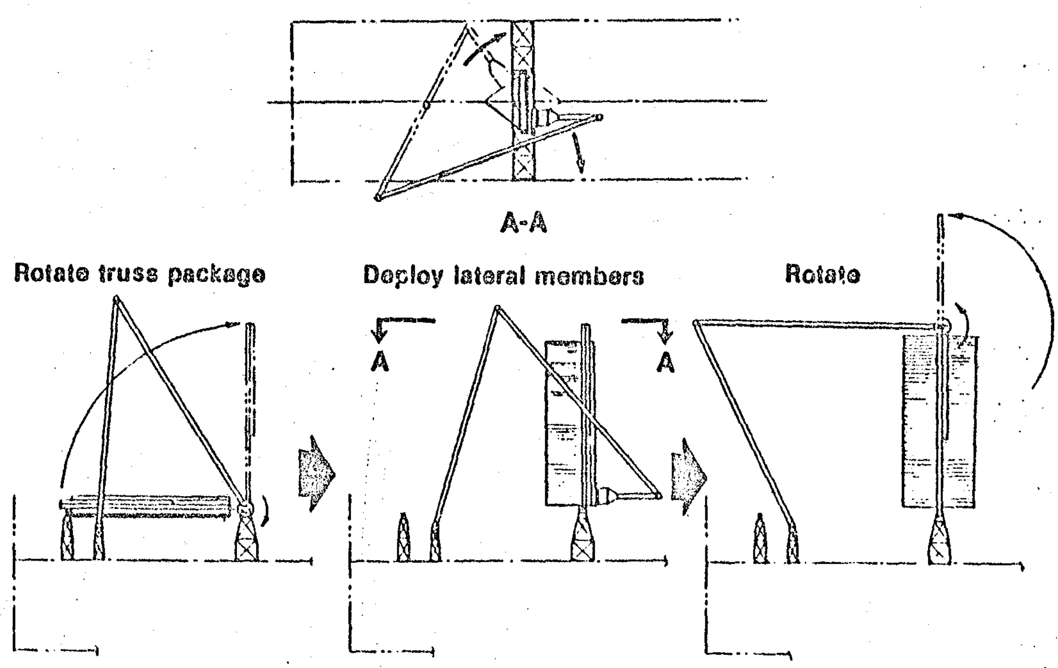


Figure 3-1. RMS for Truss Deployment Concept

Because the size of the grapple fitting for the standard end effector is prohibitively large and costly to be installed at numerous points, concepts for attaching a drive socket wrench to a standard grapple fitting were evaluated and incorporated in the SCE preliminary design. One such concept uses the Universal Service Tool (UST) shown in Figure 3-2 (Reference 10).

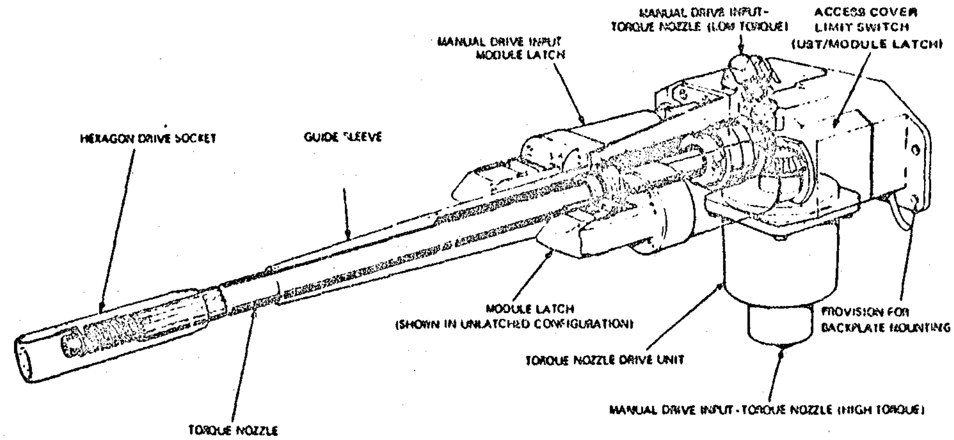


Figure 3-2. Universal Service Tool (UST)

The UST was developed by Spar Aerospace, Ltd. for the Goddard Space Flight Center (GSFC) Multimission Modular Spacecraft program. With modifications, this device could be used to perform not only RMS predeployment (and retraction) functions, but numerous installation tasks of attaching and removing subsystem modules on the structure.

An alternative concept for performing RMS predeployment operations would be to attach a straight shaft, with a socket wrench at the end, to a standard grapple fitting. This would allow simple torquing operations to be performed using the wrist roll action of the RMS. This device and the UST are considered prime candidates for RMS suitcase experiments.

Ground test and simulations of predeployment preparations and operations are feasible using the capabilities of the Manipulator Development Facility (MDF), and the SIMFAC. Such tests and simulations would be mandatory for development of the techniques, equipment, and procedures required for RMS-assisted deployment and retraction operations. However, flight test would be required to verify the total operational concept and compare the results with the ground test and simulation experience. This is because it is assumed that flight test experience with the RMS prior to the flight of the SCE will be primarily concerned with payload deployment and retrieval operations rather than the intricate specialized operations needed for space construction. Satisfactory simulated zero-g ground testing of actual RMS-aided deployment and retraction operations would be complex and expensive to perform.

3.2.3 ISSUE NO. 3 - JIGS, FIXTURES, TRACKS, AND ACCESSORIES. The handling, control, and disposition of jigs, fixtures, tracks, and accessories required to deploy and retract structures before, during and after deployment have also been addressed in the design of the basic experiment (Section 4). This issue relates directly to issues 1 and 2 because it pertains to the drives, mechanisms, supports, and controls required to support, deploy, and retract the structure.

An example of the types of devices required is seen in the Convair-developed concept for lateral and linear deployment of the selected structure concept as shown in Figure 3-3. The completely folded truss is a long flat package of hinged and solid struts held in place for transportation by a holddown and lift arm.

The lateral strut members are deployed on each side of the truss, using the holddown and lift arm. The truss is then deployed, bay-by-bay, by a motor-driven steel tape on each side of the truss, equipped with a drive latch. The drive latch reciprocates to unlatch and deploy one bay at a time while the truss is supported in the guide rails. The guide rails react applied bending moments on the truss during deployment to prevent its collapse. This concept does not include provisions for automatic retraction of the truss, as discussed in the following subsection.

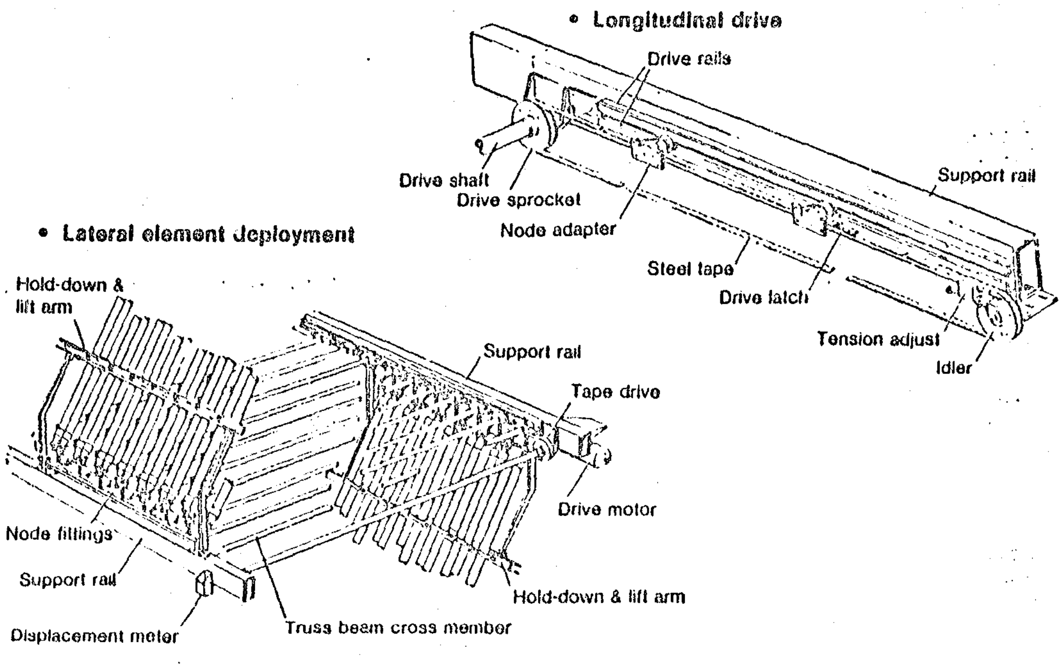


Figure 3-3. Convair-developed Truss Deployment Mechanisms and Elements

3.2.4 ISSUE NO. 4 - CONTROL OF STRUCTURAL DEPLOYMENT AND RETRACTION. Linear deployment and retraction of the structure attached to the Orbiter must be controlled to ensure the reliability and safety of the operation. This includes the ability to interrupt the operation at any time in a safe mode, provide a safe rate of deployment and retraction, prevent unacceptable loads on structural members, ensure lockup of each bay of the structure during deployment, and ensure unlocking of each bay during retraction.

Linear retraction of the truss is complicated by virtue of having to unlatch four hinges simultaneously to allow one full bay to retract. The hinge position then changes by one stacked bay height, so the hinge unlatch mechanism has to be indexed for each bay to be retracted. These operations can be accomplished by a guided carriage on each side of the beam, each equipped with two motor-driven cams as shown in Figure 3-4. The carriage can be indexed to place each trip-motor in position as subsequent bays are retracted. The carriage may either have its own drive mechanism, or be positioned by the primary deploy/retract drive. The support arms for the apex hinge tripmotors would be folded down along the deployment rail for stowage.

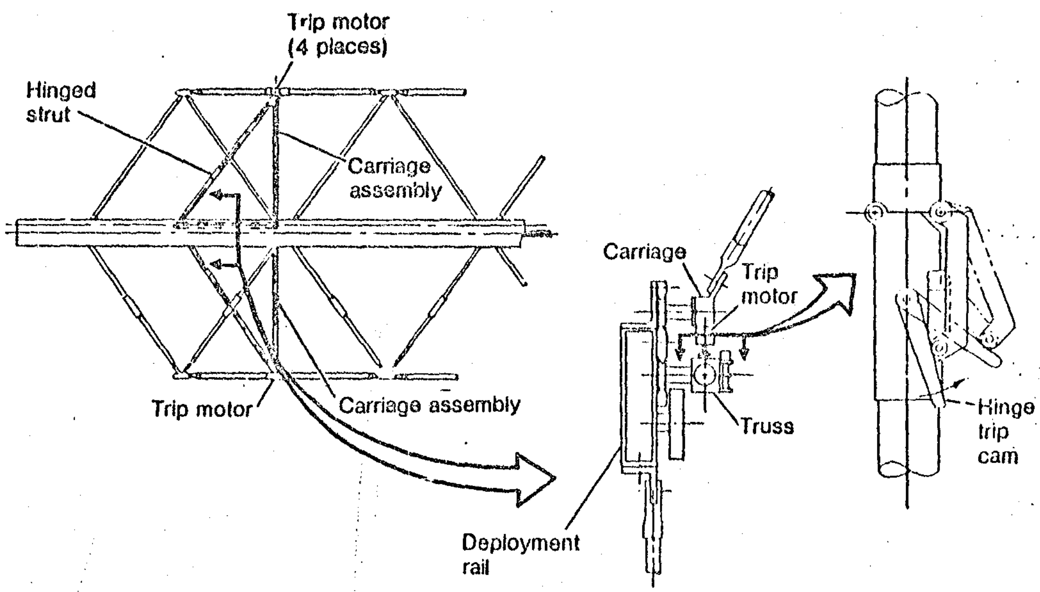


Figure 3-4. Truss Retraction Mechanism Concept

This initial concept was used to illustrate the feasibility of retracting the deployable truss. This was a major issue in deciding whether or not to recommend restowage and return of the basic flight experiment structure. The alternative would be to jettison the experiment as a free flyer or just to allow its destruction through reentry of the earth's atmosphere.

The retraction concept was further refined, simplified, and incorporated in the preliminary design of the SCE, as discussed in Section 4. Electronic controls concepts are also discussed in Section 4.

The SCE provides an early opportunity to space-test the performance and behavior of a single deployable primary structure under conditions representative of those required to construct many of the proposed large space platforms. Ground testing to simulate zero-g environment during controlled deployment and retraction would not only be costly, but, assuming the tests were conducted in an air bearing support facility, the total three dimensional loading and vibrational behavior of the truss could not be completely simulated.

3.2.5 ISSUE NO. 5 - INPROCESS QUALITY VERIFICATION AND CONDITION MONITORING. The ability to verify the condition of a space structure before and during deployment has major safety implications. Failure of structural elements due to damage sustained during launch or failure of a section of a long truss to lock-up in the deployed condition could result in toppling of the truss column and possible damage to the Orbiter.

Techniques to consider for inprocess quality verification and condition monitoring include automated, passive, and visual. Automated techniques through the use of sensors and controls can be costly and complex to implement. Passive techniques such as mechanical interlocks and go-no-go devices can create more problems than they prevent.

Since all critical truss deployment actions take place in or near the Orbiter cargo bay, the use of visual techniques to verify condition can be used extensively.

The Closed Circuit Television (CCTV) system in the cargo bay will allow fore and aft viewing of the predeployment and deployment processes. Closer surveillance and inspection of the structure and structureal mechanisms can be performed with the wrist CCTV camera on the RMS.

Simple automated techniques for condition monitoring can be provided in the deployment drive and control subsystem. Position feedback in the carriage drive mechanisms (Section 4) can be used to verify full extension of each truss bay. If necessary, the deployment carriages could be programmed to provide a "tug" on each truss bay at the end of each deployment stroke.

While the above quality verification techniques can be fully tested and verified on the ground, they would of necessity become an integrated part of the SCE, and an important element of the overall demonstration.

3.2.6 ISSUE NO. 6 - CHECKOUT, MAINTENANCE, AND REPAIR. LSS checkout, maintenance, and repair techniques, equipment, and procedures cannot only be tested as part of the SCE, but must be developed to some extent to ensure the success of SCE.

Tests in this area should be concerned with EVA operations to perform contingency maintenance and repairs. They should also include tests of checkout procedures where significant cost advantages can be achieved by employing manual techniques.

Potential candidate checkout, maintenance, and repair tests or experiments include:

a. Performance of structural alignment checkout with optical aids.

b. Isolation and repair of electrical harness faults.

c. Component replacement.

d. Structural damage repair.

The number and extent of tests planned depend on time available as discussed in Section 7. Flight crews will, however, require ground training in the techniques required for contingency repairs to the SCE structure and equipment.

3.2.7 ISSUE NO. 7 - ATTACHMENT OR JOINING OF MAJOR STRUCTURAL ELEMENTS AND SUBSYSTEM MODULES. Large space platform construction will require the joining of major structural elements and the attachment of subsystem elements to major structural elements. These joining and attachment operations will also require interfacing of subsystem power, control, data, and fluid conduits.

Techniques and equipment for docking, alignment, and mating are currently in the conceptual stage of development and are not expected to be available for inclusion in the SCE. Development of special docking, alignment, and mating devices as part of the SCE is prohibited by cost constraints. Stowage space and mission timeline limitations also prohibit inclusion of major hardware elements and tests in the basic flight experiment. However, later flight experiments reusing elements of the basic experiment could be performed.

To satisfy the need for early test of major structural attachment and joining techniques, concepts for two optional experiments were prepared. These concepts are discussed in Section 7.

3.2.8 ISSUE NO. 8 - INSTALLATION OF SUBSYSTEM EQUIPMENT. Construction of major space platforms will require the capability to install subsystem equipment before, during, and after deployment of the structure. As pointed out in Subsection 3.2.1, the preinstallation of subsystems hardware such as conduits, interface mechanisms, and subsystem modules will minimize on-orbit assembly; however, it will not always be feasible or practical to attempt to preinstall all subsystem hardware.

Thus, techniques must be devised to install subsystems hardware during or after deployment. Installation during deployment is facilitated by controlled deployment, allowing the process to be stopped and held in a safe mode at any stage of deployment. This technique is most convenient because it allows a fixed work station position to be used for numerous installation operations. When deploying from the Orbiter, it also allows this work station to be maintained within reach of the RMS. The RMS can then be used effectively to transport equipment from the cargo bay to the work station.

Where installation of hardware is not feasible until the structure is fully deployed, work may have to be performed at positions remote to the Orbiter. Suitable configured structures such as that selected for the SCE will allow the astronaut to traverse the length of the truss beam by manual power; i.e., hand-over-hand, while tethered to the Orbiter with a long lanyard. This type of operation would limit the size or amount of equipment that could be transported to the remote work station by the astronaut.

The manual installation of large or numerous pieces of equipment at remote locations on the structure should be minimized in LSS design because it represents a major expenditure of time and human energy that might better be spent in other operations. However, when necessary, the MMU provides the capability to transport man and equipment to remote locations. With its proposed "cargo worksite attachment device" it can be used to transport large pieces of equipment and temporarily attach a crew member to a worksite for specific tasks (Reference 4). Utilization of an MMU for the SCE will depend on its development status and mission timeline constraints.

A concept was developed for a universal equipment attachment module which would provide a means of interfacing both man and equipment with the truss structure. The device as shown in Figure 3-5 would attach simply and quickly to the structure at any location near a node fitting by manually activating a handcrank or handwheel or a toggle handle.

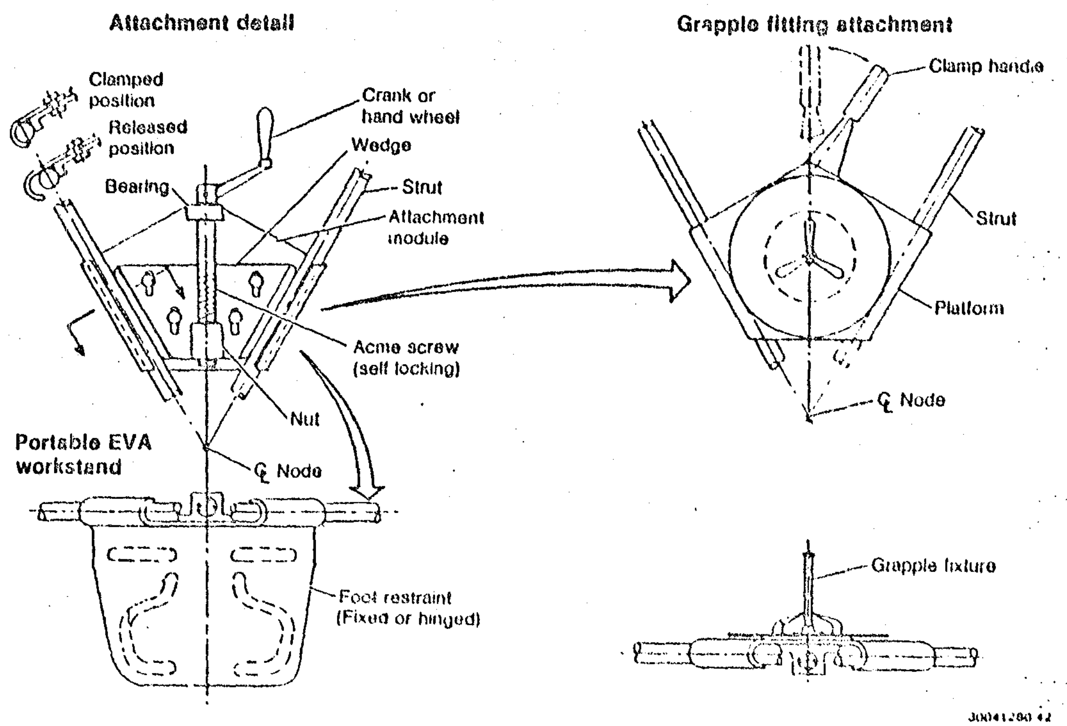


Figure 3-5. Universal Equipment Attachment Module Concept

Besides providing a base for attaching equipment modules, it could be used as a portable work station by attaching a foot restraint plate. For handling and maneuvering of truss members with the RMS, a standard grapple fixture and target could also be quickly installed with this device.

The installation of standoffs on the structure is most effectively accomplished by attaching a tripod to the truss as illustrated in Figure 3-6. A number of quick connect joints which can be integrated into the truss structure have been developed by NASA.

The quick-connect socket joints can be preinstalled on any of the truss node fittings. Struts of a number of varieties (e.g., hinged, telescoping, or nested) can then be installed at desired locations. These struts can support such items as antennas, membranes, arrays, feed modules, sensors, and propulsion modules. The sockets can also be used to make truss-to-truss joints to form planar structures or to attach support arms for major platform elements such as large reflectors and berthing modules.

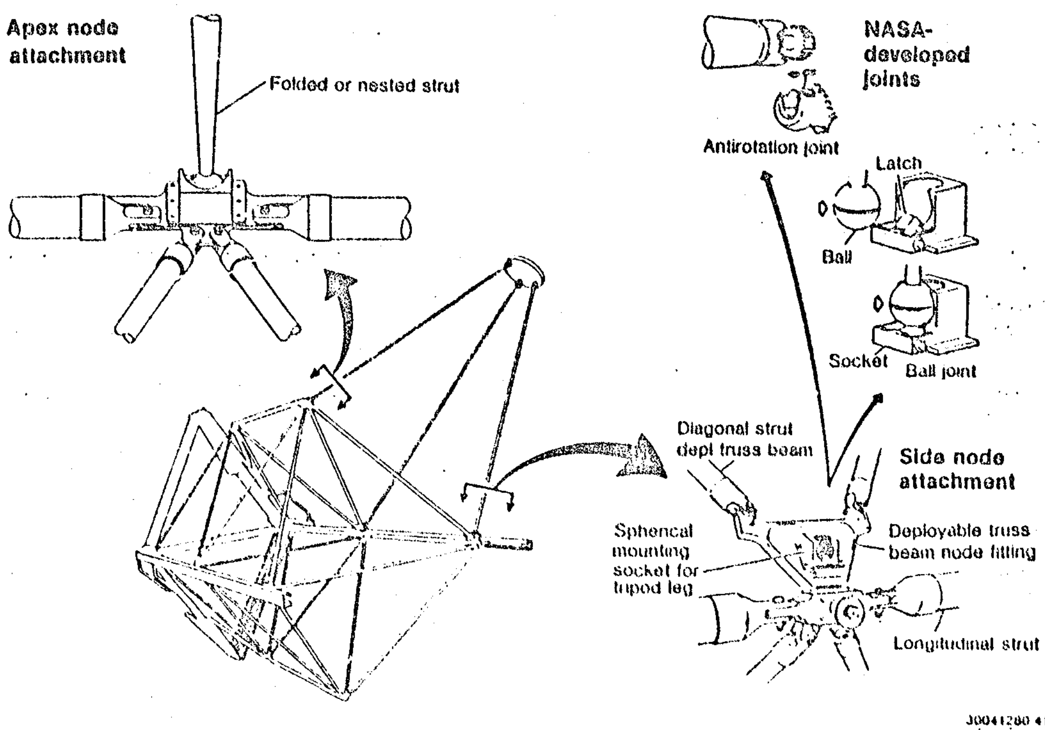


Figure 3-6. Structural Attachment Concept for Tripod Outrigger

The installation of modular subsystem elements on the side of the structure may be accomplished as shown in Figure 3-7. With three universal attachment modules and a special equipment support, electronic packages can be quickly installed and connected to preinstalled harnesses on the structure.

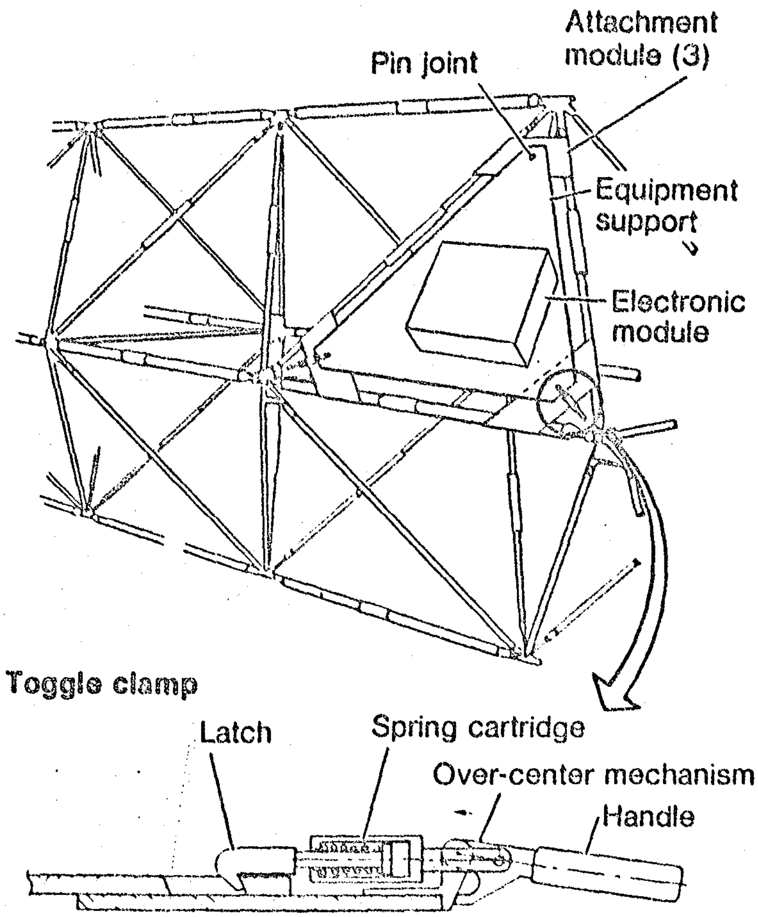
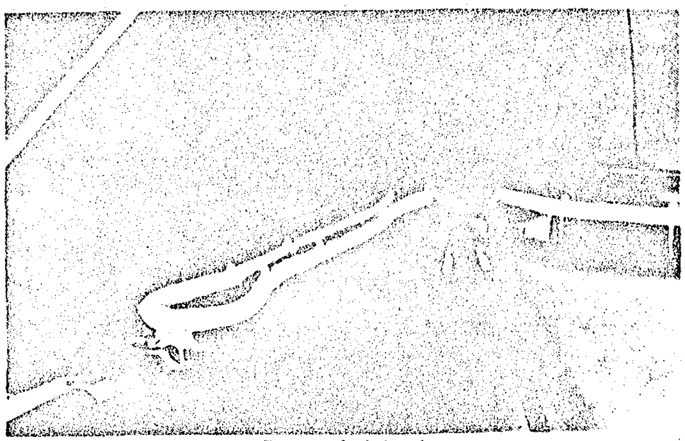
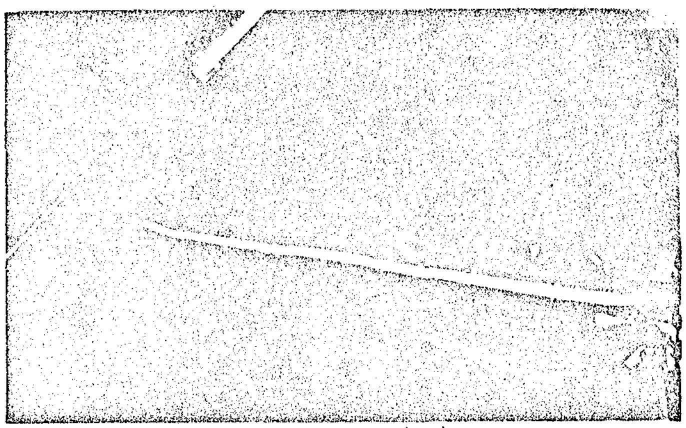


Figure 3-7. Electronic Module Installation Concept

The ability to preinstall electrical harnesses without affecting the folded geometry of the deployable truss has been demonstrated as shown in Figure 3-8. This also provides an opportunity to evaluate candidate cables and connectors for LSS applications, including techniques of securing loose connectors and manually connecting harnesses to electronic modules.



Strut folded



Strut extended

Figure 3-8. Preinstalled Conduit Demonstration

3.2.9 ISSUE NO. 9 - COMBINED EVA/RMS INSTALLATION AND ASSEMBLY. The capabilities and techniques for combined EVA/RMS installation and assembly operations are essential for performing the installation and assembly operations discussed in the preceding subsections. This includes the following types of activities:

a. Transfer parts and equipment by RMS from the cargo bay to the EVA workstation.

b. Pick up and hand off parts and equipment.

c. Position and hold parts and equipment in place during manual attachment.

d. Support and position construction aids such as the open cherry picker, tool holders, lighting fixtures.

e. Engage, maneuver, and position structural elements for manual attachment.

f. Provide a handhold support if required.

A proposed EVA/RMS experiment concept is illustrated in Figure 3-9. In this case, a simulated antenna feed module is being attached to the deployable truss structure. A portable workstation is provided by a universal attachment module (Figure 3-5) with foot restraints. Use of the truss members for handholds is permissible for the type, size, and strength of structure selected. While the experiment appears simple, it includes many of the operations and techniques previously discussed.

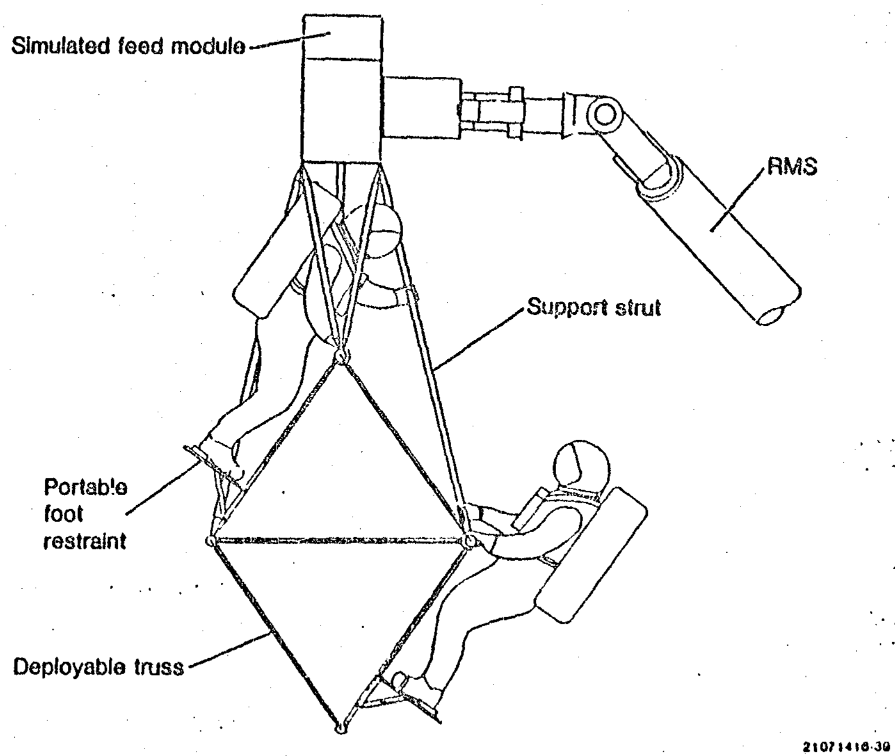


Figure 3-9. EVA/RMS Construction Operations Test Concept

3.2.10 <u>ISSUE NO. 10 - APPLICATION AND EFFECTIVENESS OF SPECIAL RMS END EFFECTORS</u>. Numerous special end effectors have been proposed for the RMS (Reference 11). Special end effectors are generally devices which can be adapted to the end of the RMS to provide the capability to engage, hold, and maneuver pieces of hardware which do not have a grapple fixture. Adaptive end effectors such as the one proposed in Reference 8 have remotely controlled movable jaws with load sensing capability for grasping objects. Devices such as the Universal Servicing Tool (Figure 3-2) have special functions such as fastening and unfastening of the modules it is designed to handle.

The SCE provides an excellent opportunity to test and evaluate special end effectors for space construction. However, with the exception of the UST, it is anticipated there will not be special, remote controlled end effectors available for test in time to support SCE.

SCE will require some kind of RMS end piece to facilitate pickup and handoff of equipment from the cargo bay to the workstation on the structure and vice-versa. With one astronaut in the cargo bay and one on the structure, a manually operated device could be used.

Figure 3-10 shows the conceptual design of a low-cost special RMS end piece which can be used by astronauts on EVA. One side is equipped with a grapple fitting to engage the standard end effector on the RMS. The other side has adjustable jaws to grasp packages of various sizes and shapes up to 0.6m in width. The adjustment crank can be easily worked by the astronauts. This special end piece will allow the astronauts to conduct designated construction operations which entail translation of structural and subsystem components to the deployable structure.

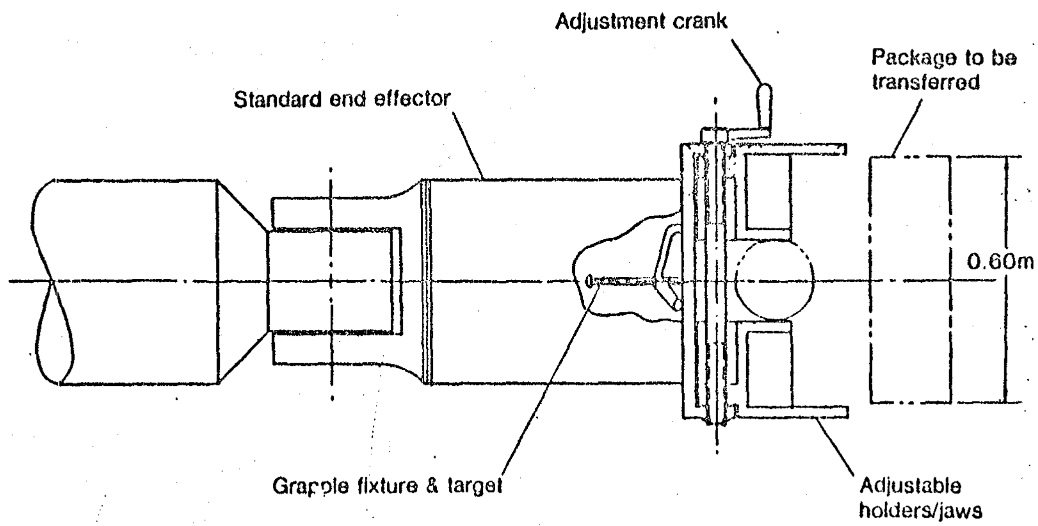


Figure 3-10. Special RMS End Piece Concept

3.2.11 ISSUE NO. 11 - ILLUMINATION AND VISIBILITY. The tests and experiments discussed in the preceding subsections for evaluating significant space construction issues rely heavily on human activities guided by human vision. Therefore, it becomes critically important that construction operations be carefully planned to ensure acceptable illumination levels for both direct and CCTV aided visual observations.

The natural illumination environment for the Orbiter in LEO is continually changing from bright to dark. Orbiter-provided lighting during the eclipse phase of the orbit should be held to a minimum to avoid excess use of Orbiter power (Reference 4). Also the Orbiter orientation must be selected to avoid direct visual contact with the sun. These considerations make inspace tests essential for evaluating illumination and visibility effects on space construction operations.

3.2.12 ISSUE NO. 12 - SEPARATION AND RELEASE OF STRUCTURE FROM THE ORBITER. The construction and assembly of LSSs will necessitate having a means of controlled separation and release of deployed structural elements from the Orbiter. This requirement satisfies both operational and safety considerations.

Assembly operations may be performed by deploying a structural beam while attached to the Orbiter to facilitate installation of subsystem hardware during incremental deployment. When fully deployed, the structure may be maneuvered to some orientation and position-stabilized by inherent or active damping; engaged by the RMS, PIDA, or HAPA; separated from its Orbiter attach points; translated; and released. The ability to separate a large attached structure from the Orbiter at any time is also a safety requirement. In the event the structure fails to fully deploy, or other emergencies cause the mission to be aborted, the structure must always be jettisonable so that the cargo bay doors can be closed.

Restowage and return of the SCE is a primary design goal. However, jettison capability must also be provided. This provides the option to release the SCE as a free flyer for extended space flight experiments.

3.2.13 ISSUE NO. 13 - ATTACHMENT OR BERTHING OF STRUCTURE TO ORBITER. There are a number of proposed techniques and pieces of equipment to allow space structures to be reattached to the Orbiter. These include:

a. Berthing of Orbiter to a large structure for servicing and support operations. This requires Orbiter control techniques to approach the structure, RMS engagement of the structure to allow the RMS to draw the Orbiter and structure together, and a berthing latch interface mechanism (BLIM) to mate and secure the Orbiter and structure together (References 4 and 8).

b. Docking of the Orbiter to a large structure requiring maneuvering the Orbiter into contact with the structure so as to engage a docking mechanism to secure the two bodies together.

c. Individual structural elements such as truss beams may be maneuvered and reattached to the Orbiter using the RMS, PIDA, and/or HAPA. This would be used where several pieces of structure may be joined together using the PIDA or HAPA as a holding fixture while positioning together pieces of structure for assembly with the RMS.

The development plans for berthing and docking devices as well as the PIDA and HAPA are not firm. This will probably preclude their use as part of an early SCE.

3.2.14 ISSUE NO. 14 - HANDLING AND POSITIONING OF STRUCTURE. The use of the RMS to handle a large flexible structural element needs further evaluation. It is currently restricted to handling payloads with natural frequencies of 5 Hz or greater, unless simulations show acceptable system response. The SCE could provide an early opportunity to perform a test of RMS/flexible structure interactions by using the RMS to engage and maneuver a length of deployable truss structure.

3.2.15 ISSUE NO. 15 - RESTOWAGE OF DEPLOYABLE STRUCTURES AND EQUIPMENT. Deployed structures for LSS applications would normally not be retracted. Once an LSS is put in operation, it will continue to function in space for many years. For systems in GEO, a policy for disposal of spent LSSs remains to be determined. For systems in LEO, disposal through reentry of the earth's atmosphere could create potentially dangerous earth impact problems similar to those experienced with Skylab. Thus, it is conceivable that LSSs would eventually be dismantled, restowed in the Orbiter, and returned to earth for disposal. Early deployable systems may be fully deployed in LEO for checkout. These systems could have retraction capability to allow the spacecraft to be refolded for transfer to a higher orbit.

Regardless of ultimate requirements to restow space structures, the SCE should address this issue because of the benefits to be gained by having a retraction and restowage capability. These include:

a. Postflight evaluation of experiment hardware to identify real and potential problems resulting from inspace operations.

b. Ability to retract the structure during crew sleep periods to preclude continuous watch and/or monitoring of the deployed structure as a safety precaution.

c. Ability to quickly back out of test conditions in case of emergency.

d. Ability to return the structure to any partially deployed condition for further tests and evaluations.

e. Ability to perform structures, dynamics, and controls tests in a complete sequence, then retract the truss to perform a series of construction operations tests by installing hardware as the truss is incrementally deployed. This allows a series of hardware installation tasks to be performed within reach of the RMS, which appears to be an efficient construction technique.

f. Recovery of flight-test hardware for reuse on subsequent system flight experiments.

3.2.16 ISSUE NO. 16 - ORBITER-INDUCED DYNAMIC EFFECTS. Maneuvering, pointing, orienting, and holding the position of large space structures attached to the Orbiter will normally be accomplished with the Orbiter Vernier Reaction Control System (VRCS). When operated, the VRCS will apply torques which will induce vibrations in the deployable structure as well as the RMS during construction operations. The frequency of the VRCS pulse rate may at times approach the natural frequency of the deployed structure or the RMS. For control of very large attached structures, primary RCS thrust may be required. There is also concern that a VRCS thruster may not shut off due to a system failure. In this case a primary RCS thruster could be commanded to fire to correct for the failed thruster. The higher torque created by these conditions could damage the structure. Both normal and contingency Orbiter-induced dynamic effects on structure, deployment mechanisms, construction equipment, and operations must be evaluated to ensure safety and acceptable performance of Orbiter-supported LSS construction operations.

The maneuvering and attitude control requirements of large space structures attached to the Orbiter depend on construction and mission operations. This includes, for example:

a. Orientation of the structure with respect to the sun and earth to enhance visibility during deployment and construction.

b. Pointing a long structure to facilitate LSS subsystem experiments; e.g., antenna elements, earth sensors, inertial sensors, etc.

c. Orientation of a structural element in relationship to a space platform in preparation for separation from the Orbiter and joining to the platform.

d. Orientation of a long structure in a gravity-gradient-stabilized attitude in preparation for release.

There is little concern that VRCS pulse rates would excite the natural frequency of the structure to a sufficient degree to cause damage. Such frequencies can be avoided by programming of the autopilot. If they should occur for brief periods, only minor structural excitation would occur. It has been shown experimentally that the amplitude of structural deflection builds up rather slowly, such that several minutes of continuous excitation are required to reach significant amplitudes. Finally, there is the consideration of providing active damping on Orbiter-attached structures to prevent all adverse vibration effects.

High bending moments on Orbiter-attached structures due to primary RCS thruster firings have more serious implications. In the case of a long attached structure with a large moment of inertia (e.g., feed mast with a large reflector or feed module at the tip), the structure will have to be designed to withstand these loads without failure. This will penalize the structure in terms of size and cost. Alternatives to consider include:

a. Modify the Orbiter RCS to preclude adverse failure modes and/or PRCS operation.

b. Provide a load-control mounting interface for attached structures.

c. Perform structural deployment and assembly free of the Orbiter.

It is reasonable to expect that experience with and improvement of the Orbiter RCS over the years of Shuttle operations will minimize the probability of adverse failure modes during space construction operations. Major redesign of the RCS is not likely to occur unless dictated by major program considerations. However, design policy with respect to attached structures may be modified with time.

Bending loads on attached space structures can be attenuated at the mounting interface through the use of spring mounts or load stabilization actuators; e.g., spring-loaded dashpots. The Orbiter control implications of flexible mounting of the SCE structure are discussed in Subsection 6.3.4.

Deployment and assembly of large space systems free of the Orbiter is of greatest advantage, if little or no assembly and installation work is required to complete the system. The MMU can be used to transport men and equipment to the structure after deployment; however, the advantages of remaining attached to the Orbiter are lost. Consideration of other factors such as orbital transfer loads and durability provisions to protect the structure from human contact damage may minimize size and strength reductions gained through free deployment and assembly.

3.2.17 ISSUE NO. 17 - MEASURED VERSUS PREDICTED STRUCTURAL DYNAMIC CHARACTERISTICS. An Orbiter-attached space structure experiment will allow Orbiter-induced structural dynamic effects and their control to be carefully measured and evaluated prior to undertaking space construction projects of major scope. More importantly, it will allow measured behavior to be compared and correlated with that predicted by ground tests and simulations.

Because of the cost, complexities, and uncertainties of performing full scale structural dynamics tests on the ground, the dynamic performance of structures in space will be predicted by analytical techniques. These computer analyses will be augmented by measured characteristic parameters derived from component and subassembly tests; e.g., weight, modulus, stiffness, damping, etc.

Initial verification of ground test and simulation techniques of predicting space structure dynamic performance is best performed by flight test of a simple structural member such as a truss beam. This type of structure is simple to model and free of variables introduced by major hardware and structural attachments.

After an initial test of a bare beam, hardware installations can be made and dynamic performance tests repeated. A special option would be to separate the beam from the Orbiter and perform free-free dynamic performance tests. This option is considered further in Section 4. Dynamic testing considerations for a simple attached truss beam experiment are discussed in Subsection 6.1.

3.2.18 ISSUE NO. 18 - STRUCTURAL DAMPING. Damping and control of vibration in large flexible space structures is usually considered an operational function. The stabilization of precision reflector surfaces and the supports for devices which must be accurately pointed or undisturbed is essential to the operation of most LSSs. It becomes a construction consideration, however, when quiescence of large structural elements or subassemblies must be maintained to ensure their safe handling, maneuvering, and joining in space.

Ground tests to measure inherent structural damping characteristics and performance of active damping techniques and equipment are limited to relatively small structural models or elements. Ground tests are usually limited to 2 or 3 degrees of freedom in simulated zero-g. Damping effects due to air are eliminated by performing tests in vacuum chambers. This further limits the size of test specimens.

An early Shuttle flight experiment of a representative LSS structural truss member will be the first step to gaining data and experience in damping and vibration control. It will: (1) allow vibration modes and frequencies of interest and concern to be identified; (2) provide data on inherent structural damping and non-linear damping effects with a much greater degree of certainty than can be achieved through ground tests; and (3) allow components and techniques for complex LSS active control to be evaluated.

Options for additional space tests could include: (1) expanding the configuration of the test structure to evaluate planar structures; or (2) equipping the test truss with necessary communications, power, and control packages and releasing it from the Orbiter to perform free-free beam tests. These options are contingent upon cost and stowage space constraints.

The recommended approach for damping and control of the SCE is discussed in Subsection 6.2.

3.2.19 ISSUE NO. 19 - STRUCTURAL RATTLE AND BACKLASH EFFECTS. Deployable space structures are foldable/expandable mechanisms. The mechanisms may employ hinged joints, pivot joints, sliding joints, stops, locks, latches, springs, bearings, and gears to connect load-carrying elements such as struts, rods, cylinders, cables, and beams together to form a rigid structure when fully expanded. At each movable connection there is a tiny clearance between mating elements which may be on the order of hundredths or thousandths of a millimeter, depending on the manufacturing precision. Multiplied over the length of a long deployable structure (50-100m) which has thousands of these tiny clearances built into it, the result will be some finite longitudinal, lateral, and torsional free-play (rattle) due to accumulated clearances (backlash).

The design of future LSSs will require more insight into the effects of rattle and backlash than is currently known. Significant areas of concern include:

a. Precision pointing and surface accuracies may be severely limited in some cases unless zero backlash is achieved. This poses some very complex technology and design problems if foldable structures are to be widely used for LSSs.

b. Rattle and backlash contribute to the nonlinear damping characteristics of structures. For nonprecision structures, the limits of rattle and backlash and their effect on damping of LSSs need to be understood to design adequate structural damping systems.

c. Structural joint clearances and design have a significant impact on the cost and produceability of deployable structures. Knowledge of how much backlash can be tolerated is required to establish cost effective designs.

There appears to be no simple approach to this problem. Inspace test of a representative LSS structural truss is essential to obtain accurate measurement of rattle and nonlinear damping characteristics. It would be of greater value if backlash could be varied so that a range of rattle could be evaluated. Providing variable clearance in a few joints could be considered; however, this needs to be evaluated in detail.

Development and test of a zero backlash structure is another alternative. Inspace tests would then be performed to verify the effectiveness of the design and measure its inherent nonlinear damping characteristics. This approach would not be cost effective because of the high cost to produce a zero backlash structure, the fact that zero backlash could be verified by ground testing, and little data would be obtained to support the design on nonprecision structures.

For the SCE, perhaps the best approach is to test a structure manufactured to reasonably close (affordable) fits and tolerances, which would be considered adequate to fulfill general purpose construction needs. The results of this test would then be analyzed to determine the need for greater precision.

3.2.20 ISSUE NO. 20 - STRUCTURAL THERMAL EFFECTS. An LSS in LEO will be subjected to the continuously changing thermal radiation environments of incident solar, earth thermal, and earth albedo as it travels through each orbit. Lattice structures such as truss beams have multiple slender elements with earth-facing surfaces and sun-facing surfaces. Sun-facing surfaces may also be partially shadowed by up-sun structural members. The heat flux and temperature distribution, thus, varies over the surface of each truss element with position and time.

The thermal response of individual truss elements at discrete points on the element can be predicted by computer techniques (Reference 24). This permits thermal mapping of each element, which in turn can be input to structural analysis programs such as LASS (Reference 25) or NASTRAN to predict the effects of thermal expansion and contraction on the overall structural shape and dynamic response.

Thermal expansion and contraction of structural elements can result in deflecting truss beams from their neutral alignment. Sudden transitions in thermal radiation environment (as from earth albedo to direct solar) can create thermal shock effects. These deflections and/or sudden disturbances can seriously impact control of pointing and surface accuracies.

The ideal solution to this problem is to use structures with insignificant (near zero) net coefficient of thermal expansion (CTE). This approach has been demonstrated on small spacecraft structures such as the HEAO-B optical bench. Using graphite/epoxy composite materials with standard pseudo-isotropic layup techniques produced a structure with a CTE within $0 \pm 10^{-7}/°F$. Lower CTE limits are possible using special layup/compounding techniques. CTEs of metallic fittings and graphite composite tubular elements can be matched to produce net zero CTE structural elements for truss beams (Reference 26).

By using a near-zero CTE structure for the SCE, thermal behavior can be accurately predicted using existing techniques augmented by thermal/vacuum ground tests of structural fittings and members. This will preclude use of expensive instrumentation for thermal mapping of the flight structure. Thermal/dynamic response characteristics can be verified using instrumentation provided for other structural dynamic characterization testing. Thermal deflection measurements would confirm predicted performance. Thermal deflection, however, may be an order of magnitude less than the free play (rattle) in the structure, and attempting to measure it could be futile.

3.2.21 <u>ISSUE NO. 21 - STRUCTURAL INERTIA AND VIBRATION EFFECTS ON ORBITER CONTROL</u>. The preceding discussions of space construction issues have pointed out the importance of the role played by the Space Shuttle Orbiter and the advantages of performing deployment, assembly, and installation operations on large structures attached to the Orbiter. Using the Orbiter to point, maneuver, and position large flexible structures with large moments of inertia raises several questions.

a. What are the effects on the Orbiter DAP?

b. What are the control limits of the DAP and how can they be defined?

c. What needs to be done to the attached structure to enhance DAP control and mission safety?

This study has provided preliminary answers to these questions, as discussed in Section 6. Tests have been defined in Section 7 to allow the DAP limits to be approached and measured in a safe, conservative manner. These tests will determine the effectiveness of DAP simulation analyses and establish an important data base for future LSS construction operations design and planning.

3.3 RECOMMENDED SCE TESTS AND EXPERIMENTS

The identification and evaluation of candidate tests and experiments concepts led to the selection of the operations tests and evaluations summarized in Table 3-2. These recommended experiments fall in the four major categories indicated.

Basic applicability indicates that all experiments can be performed in conjunction with the basic flight experiment. Suitcase applicability indicates additional hardware, not included in the basic experiment, is required to perform that experiment.

Items in the category of "later flight experiment candidates" are all considered excellent test candidates which are unavailable for early flight test due to current development plans.

3.3.1 EVOLUTIONARY SCE OPTIONS. The list of selected experiments suggests a number of optional approaches to accomplish those and future experiments. Figure 3-11 indicates an evolutionary approach with up to 5 options. These are described as follows:

a. Option 1 would limit the scope of the first flight experiment to a number of EVA and RMS suitcase experiments. A short segment of deployable structure could be used to facilitate installation and assembly tests. Since the structure would not be subject to dynamic evaluation tests, early prototype structures such as the General Dynamics Deployable Truss (Figure 3-12) might be used to minimize development cost.

b. Option 2 would be to fly the basic experiment without EVA suitcase experiments. This option would perform the structures and dynamics tests and most of the RMS tests and evaluations. EVA would be used only as a contingency measure.

c. Option 3 would be to conduct all of the EVA/RMS and structures and dynamics tests in a single mission. This would be more cost effective than performing Options 1 and 2 separately. It is a better system approach to evaluating space construction activities because it includes the broad scope of construction activities that will be required to support a major space construction project.

Table 3-2. Selected Operations Tests and Evaluations

Early Flight Experiments	Operations Tests & Evaluations	Applicability	
		Basic	Suitcase
RMS/standard end effector	Deploy & retract truss	•	
	Pickups & handoffs (special end piece)	•	•
	Position & attach modules	•	•
	Surveillance & inspection	•	
	Engage/maneuver/position truss	•	•
	Assess illumination/visibility	•	
EVA	Install structural elements/rigging	•	•
	Install subsystem elements	•	•
	Assess joints, couplings, connectors	•	•
	Repair/maintenance operations	•	
	Assess portable work aids/fixtures	•	•
	Assess illumination/visibility	•	
	Assess RMS effectiveness	•	•
	Inspection/verification/checkout	•	
Later flight experiment candidates	Birthing latch interface mechanism	•	•
	RMS special end effectors	•	•
	Handling & Positioning aid	•	•
	Open cherry picker	•	•
	MMU	•	•
Structures & dynamics	Assess deployment/retraction techniques	•	
	Assess prewiring/harnessing/conduits	•	
	Assess hardware preinstallation techniques	•	
	Interactions with EVA/RMS	•	
	Instrumentation techniques	•	
	Rattle & backlash effects	•	
	Thermal control/stability techniques	•	
	Damping techniques	•	
	Modal measurements	•	
	Interaction with orbiter DAP	•	
	Orbiter maneuvering effects	•	
	Structural joining techniques	•	
	Structural performance/behavior	•	
	– Attached to orbiter	•	
	– Free-free mode (optional)	•	

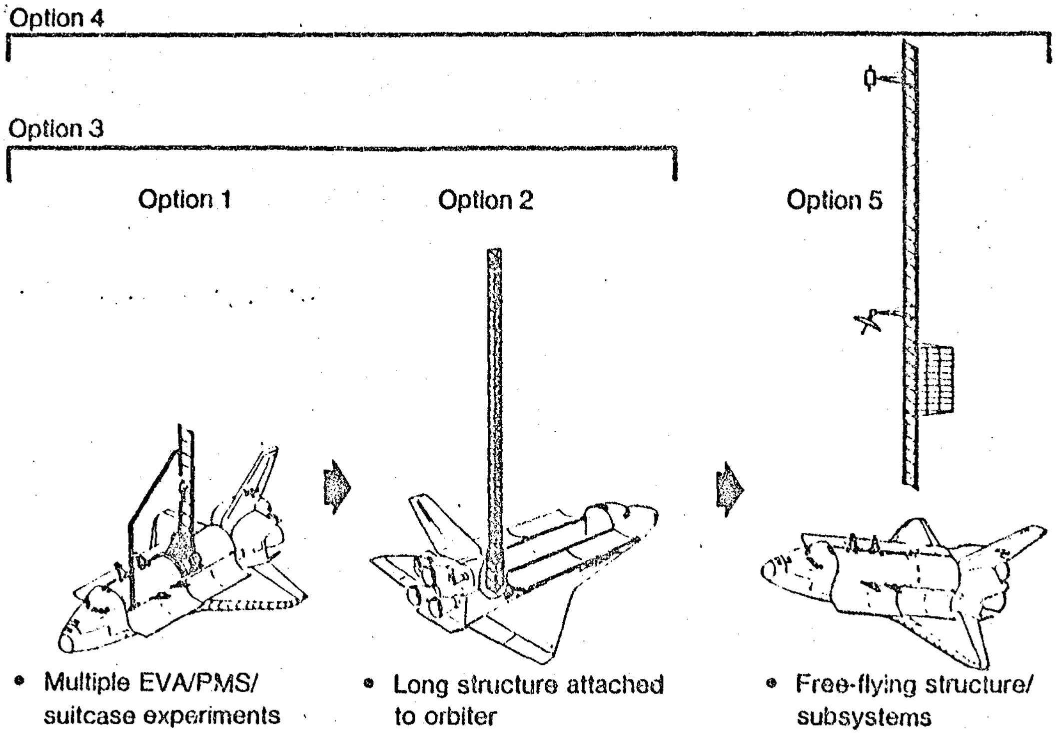


Figure 3-11. Space Construction Experiment Evolutionary Options

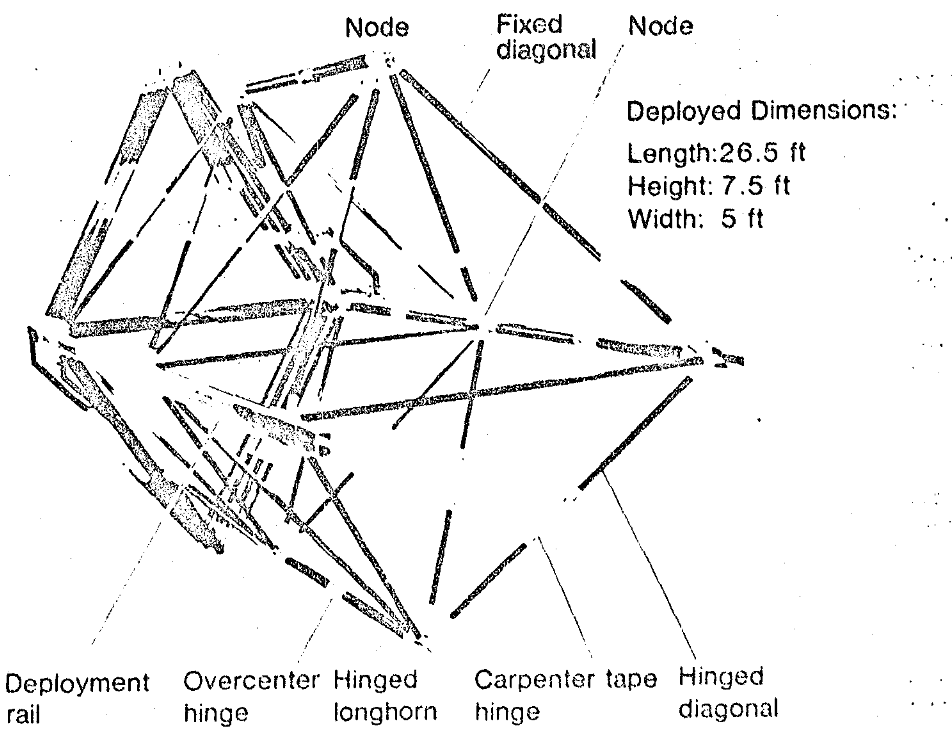


Figure 3-12. General Dynamics Prototype Deployable Truss

d. Option 4 includes all the operations necessary to perform all of the recommended SCE tests and experiments including the option to perform a free-free mode structural dynamics test. The EVA/RMS operations in this case would install the necessary equipment for free-flight testing of the structure. This option also demonstrates the structural separation from the Orbiter.

e. Option 5 is a spinoff benefit of developing the basic experiment. It is anticipated that extended flight experiments to test and evaluate LSS subsystems hardware may be required. Reuse of SCE hardware and technology will provide the capability to test a variety of LSS subsystems, either attached to the Orbiter or as free-flying experiments. It can also be reused to test more complex deployment, assembly, and controls experiments as the need arises.

SECTION 4

PRELIMINARY DESIGN

An evaluation of potential STS flight candidates was performed to identify time, space, crew, interface, and schedule constraints to the SCE. Experiment design concepts based on the selected experiments, suggested test options, and STS flight candidates were prepared and evaluated. A preferred SCE design concept was selected and preliminary design drawings prepared. These concepts, trades, evaluations, and preliminary design data are presented in this section.

4.1 STS FLIGHT CANDIDATES EVALUATION

The STS Flight Assignment Baseline per Reference 7 was analyzed to identify potential flight opportunities for the SCE. Unfortunately, these data were preliminary at the time of the analysis, and have since undergone major revision. The analysis did, however, provide good insight into the limitations and constraints of the type of missions that need to be considered.

4.1.1 EVALUATION CRITERIA. The criteria for this evaluation were:

a. The SCE will be added as a payload of opportunity on a space and weight availability basis.

b. The SCE will not displace any assigned payloads or encroach upon the allocated envelopes for access, deployment, and viewing.

c. The SCE will not interfere with the operating or deployment timelines or operating conditions required for assigned payloads.

d. Additional mission time and crew members will be added if required to perform the SCE, within the limits of a 7-day maximum mission and 6-person crew.

4.1.2 STS FLIGHT CANDIDATES EVALUATION. Of all the flight assignment plans evaluated, there were four basic types of flights that offered possibilities of SCE accommodation. These generic flights are shown in Figure 4-1 and are evaluated as follows:

a. Spacelab Pallets (STS-14, 34, and 41). The SCE may be installed forward of the pallets. These missions carry a full 6-man crew on a 7-day mission, indicating full activity devoted to the primary mission. SCE would interfere with viewing of pallets. VRCS usage restrictions were not determined. Criteria 4.1.1.c and 4.1.1.d cannot be met.

- Spacelab pallets
  STS-14, 34 & 41

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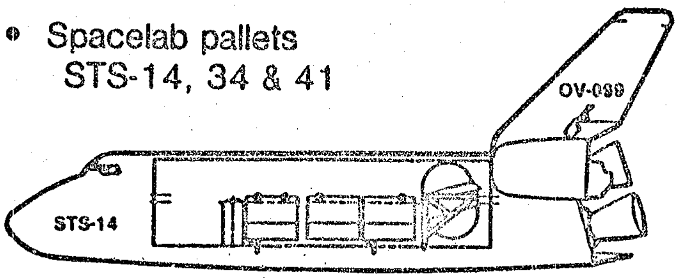
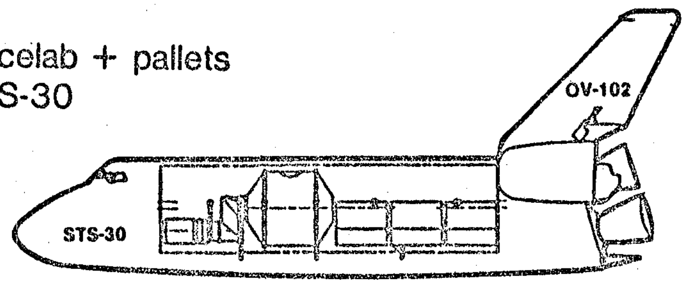
30041280-15



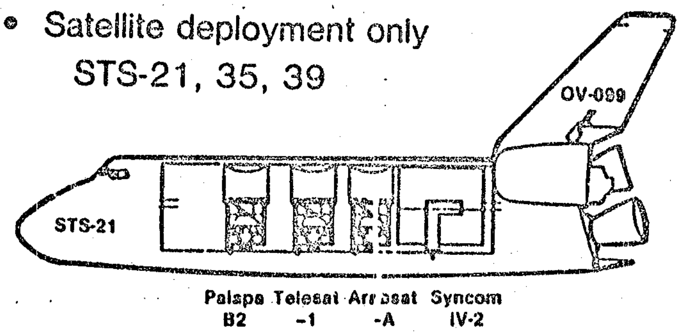
Spacelab 2

- Spacelab + pallets
  STS-30



Spacelab 6

- Satellite deployment only
  STS-21, 35, 39



- Satellite deployment + pallet
  STS-17 & 28 (MPS pallet)
  STS-24 & 32 (OSS pallet)

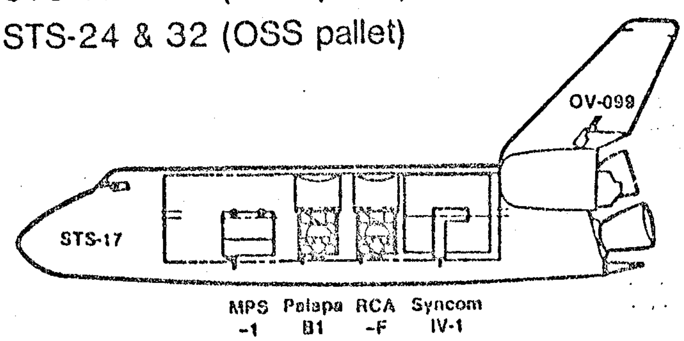


Figure 4-1. STS Flight Candidates

b. Spacelab Plus Pallets (STS-30). Only one of the six Spacelab plus pallets missions allowed an additional payload. Only space available for SCE is over the forward tunnel unless the Power Extension Package (PEP) is added to the manifest. Space above aft pallets is probably restricted. A full 6-man, 7-day mission is allocated to the primary mission. Viewing and VRCS usage restrictions not determined. Criteria 4.1.1.c and 4.1.1.d cannot be met.

c. Satellite Deployment Only (STS-21, 35, and 39). The SCE may be installed forward of the other payloads. One to five day missions are planned with 3-man crews. No restrictions after the payloads are deployed. All criteria are met; however, these missions are expected to have a high user demand for available payload space.

d. Satellite Deployment Plus Pallet (STS-17, 24, 28, and 32). The SCE may be installed forward of the pallet. The MPS pallet assigned to STS-17 and 28 has no restrictions above it, so that SCE may straddle the MPS pallet. Restrictions on RCS usage during MPS operations need to be determined. The OSS pallets assigned to STS-24 and 32 have viewing, access, and operational requirements that SCE may interfere with. There is more open work space after satellites are deployed. These are typically 7-day missions with 3-4-man crews. The MPS flights offer ample crew and EVA work time for SCE, whereas the OSS flights require more crew attention and test activity. Preliminary indications are that the MPS missions meet all criteria and the OSS missions do not meet criteria 4.1.1.c and 4.1.1.d.

4.1.3 PROGRAM CONSTRAINTS. Program constraints on the acceptability of flight candidates were investigated as part of this first-cut evaluation. A nominal development plan was laid out, assuming a FY 1983 program start, to determine the earliest probable flight date. This plan, as shown in Figure 4-2, indicated the earliest flight might be supported by the third quarter of CY 1984.

The given flight dates for the candidate missions that fall within the flight opportunity window showed that at least one of each of the types of mission candidates could be supported.

The column labeled "concept" refers to the applicability of the SCE concepts described in the next subsection to the various mission candidates.

4.2 EXPERIMENT CONCEPTS AND EVALUATIONS

Seven SCE concepts were developed and evaluated. Candidate STS flight configurations were used to determine space and envelope constraints. Two concepts were defined for use with the Spacelab 6 mission, even though the evaluation indicated it to be a poor candidate. This was done to provide more contrast between competing concepts and to illustrate a wider range of options than is seemingly available.

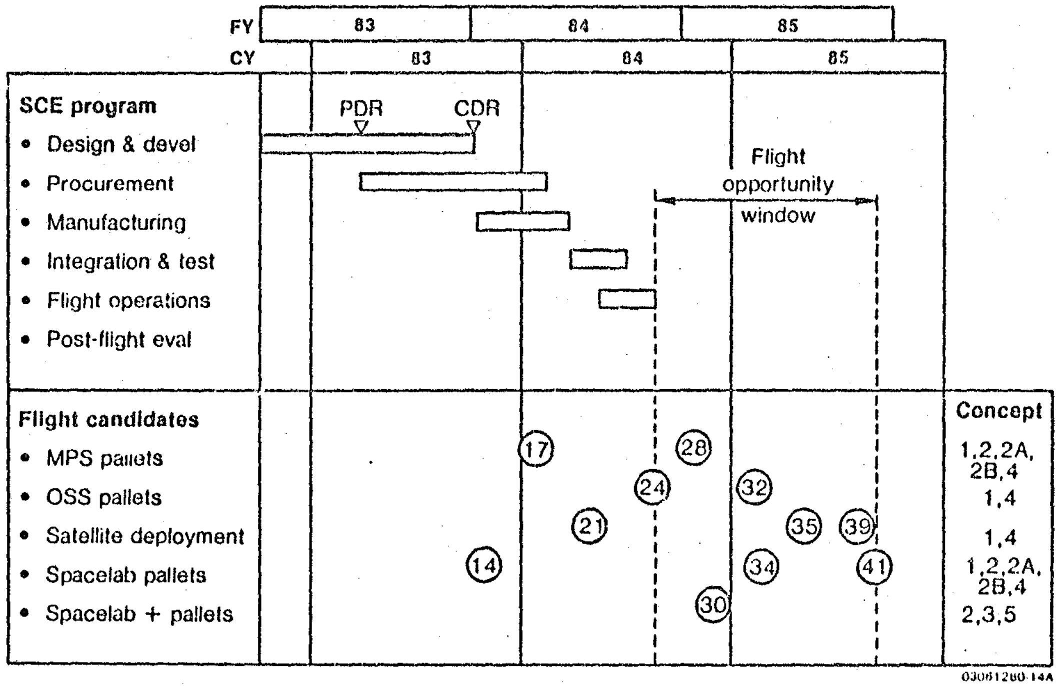


Figure 4-2. Program Constraints on Flight Candidates

4.2.1 EXPERIMENT CONCEPT 1. Experiment concept 1 is shown in Figure 4-3. The truss is stowed crosswise in the cargo bay on an equipment support pallet forward of the MPS pallet on an MPS pallet/deployable satellite mission. The support pallet is stowed forward of the MPS pallet, allowing clearance for an astronaut to get into the cargo bay in case of emergency. The length of the truss in this configuration is approximately 29m deployed. This is a marginal length for running the dynamic tests; however, it is more than adequate for the performance of the EVA/RMS experiments. This suggests the use of concept 1 as a multiple suitcase experiment only with perhaps some testing of dynamic test equipment and instrumentation.

The EVA/RMS experiment packages and hardware are stowed in the support pallet below the deployable truss. This pallet could be a standard flight support system (FSS) cradle as defined for the multi-mission spacecraft system.

This concept, as in all subsequent concepts, has the option of employing a triangular truss as a cost reduction consideration.

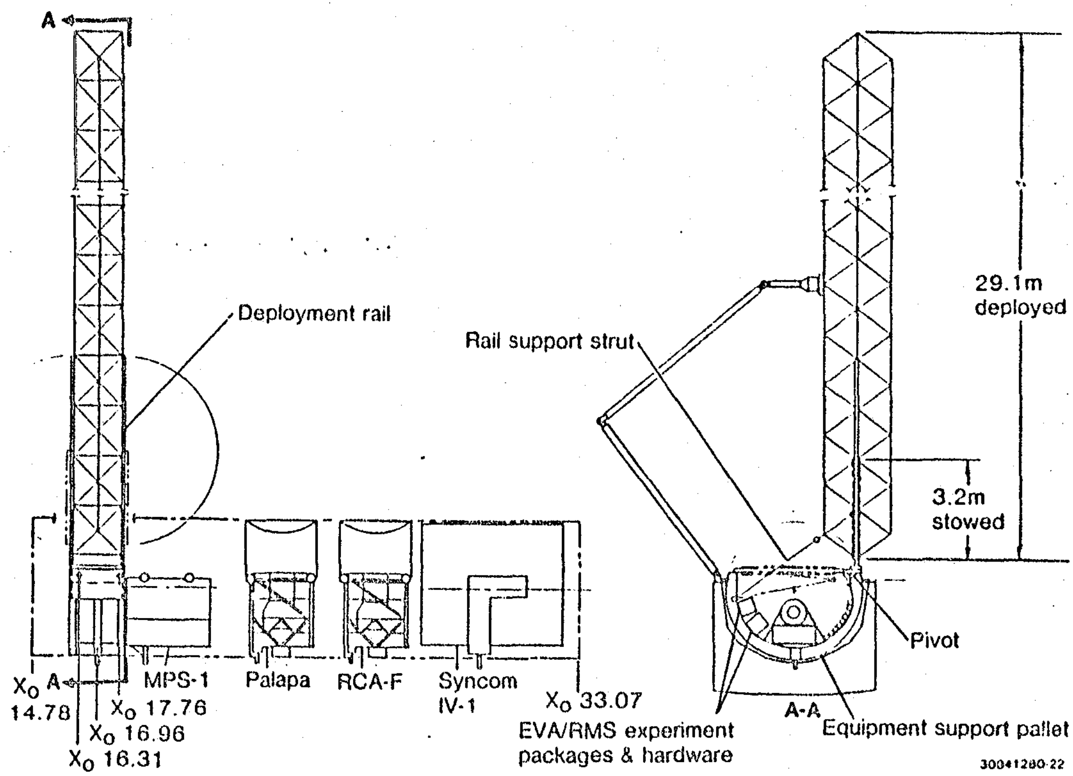


Figure 4-3. Experiment Concept 1

4.2.2 EXPERIMENT CONCEPTS 2, 2A, AND 2B. Experiment concept 2 is shown in Figure 4-4. The deployable structure is stowed above the MPS pallet on an MPS pallet/deployable satellite mission. The MPS equipment doesn't extend above Z_O 10.52; consequently, there is plenty of room for the structure. The deployed length of the structure is 42m, which is considered an adequate length to perform the desired dynamic tests. Extending the length of the stowed truss toward the aft cabin bulkhead would allow additional length of truss to be added. For the basic concept 2, only the deployable structure is carried, with no experiment packages to perform EVA/RMS experiments.

In concept 2A a support pallet can be added similar to the one in concept 1, which houses the EVA/RMS experiment packages and hardware. Clearance is still maintained for the astronauts to have access to the cargo bay in an emergency. Other equipment support arrangements may also be considered.

In concept 2B a support pallet is also added. On this pallet is placed free-free dynamics experiment hardware. The equipment is placed on the structure by EVA/RMS. After the Orbiter-attached dynamic and construction operations tests are performed, the structure is released for free flying dynamic experiments.

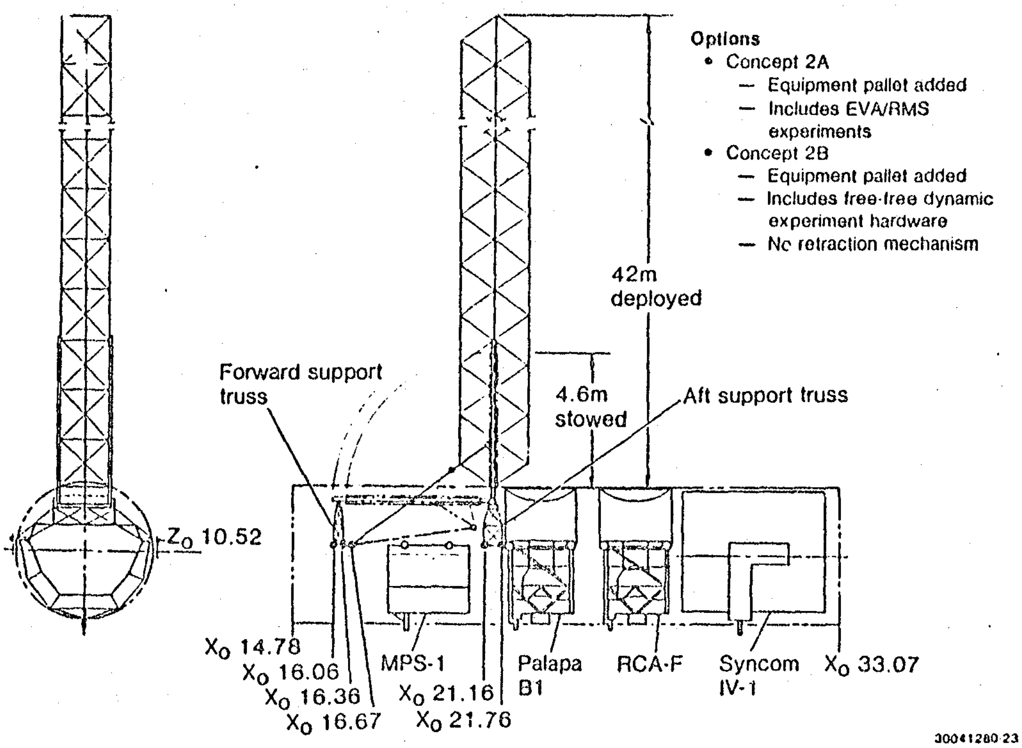


Figure 4-4. Experiment Concept 2

Figure 4-5 describes the simplifying assumptions and minimum additional hardware required to perform a free-free truss experiment. The hardware is attached by EVA/RMS after the structure has been deployed and the dynamic tests have been run while it is still attached to the Orbiter. The truss is then separated from the Orbiter and flies in formation with the Orbiter. During the additional testing, data are transmitted to the Orbiter for recording. After the free-free experiment is completed, the structure is deboosted out of orbit.

4.2.3 EXPERIMENT CONCEPT 3. An experiment concept to be flown on a Spacelab/pallet mission is shown in Figure 4-6. The structure is stowed above the pallets. The structure can reach a deployed length on this configuration of 63m. This concept carries only the deployable structure to perform the structures/dynamics tests because space limitations preclude installation of an EVA/RMS experiment equipment pallet. However, equipment to perform the EVA/RMS experiments could be carried forward on a support pallet above the tunnel to the Spacelab as an option.

Assumptions
- Can separate in gravity gradient orientation with near zero angular momentum
- No additional instrumentation & torque wheels required
- No recovery capability
- Data transmitted to orbiter only

Δ Hardware
- Battery package
- Telemetry package
- RF control package
- Separation mechanism
- Reentry package

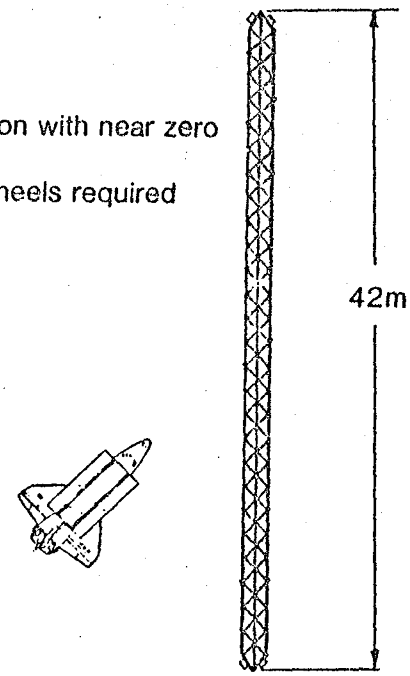


Figure 4-5. Free-free Truss Experiment Concept 2B Option 30041200-27

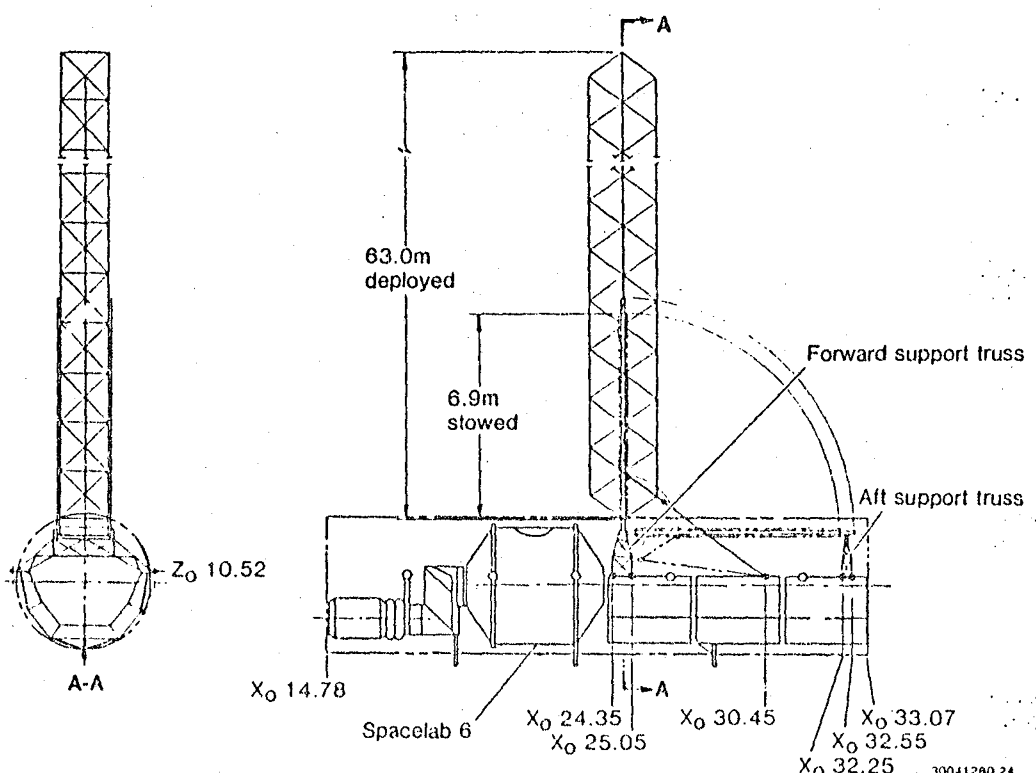


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Figure 4-6. Experiment Concept 3

4.2.4 EXPERIMENT CONCEPT 4. Concept 4 (Figure 4-7) was developed to show what the least expensive deployable truss experiment might look like. Only the structure is included in this concept supported on a support frame. No equipment for EVA/RMS experiments is included. The structure is triangular and not a diamond configuration. There is no rotation mechanism. The structure is deployed sideways from the cargo bay by the RMS. The deployed length achievable is only 22.6m, which is marginally short to perform the desired dynamic tests.

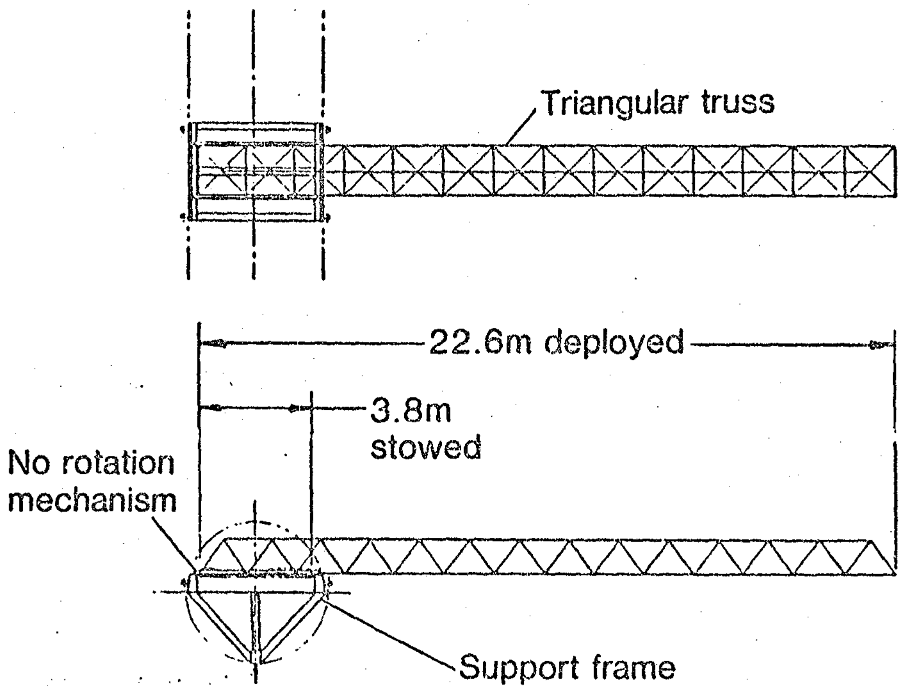


Figure 4-7. Experiment Concept 4

4.2.5 EXPERIMENT CONCEPT 5. Experiment concept 5 (Figure 4-8) is a variation of concept 3. The same deployable truss is carried aft above the pallets. In addition, another short deployable truss segment is carried forward above the tunnel to the Spacelab. There is no EVA/RMS experiment equipment with the exception of truss-to-truss attachment hardware/tools. The dynamic tests are run with the aft structure deployed. After these tests are completed, the forward truss is deployed. The forward truss is separated from the cargo bay by the RMS and attached to the aft truss to verify the operations. The forward truss is then separated from the aft truss and reattached to the Orbiter by the RMS. Both are retracted and restowed for return to earth.

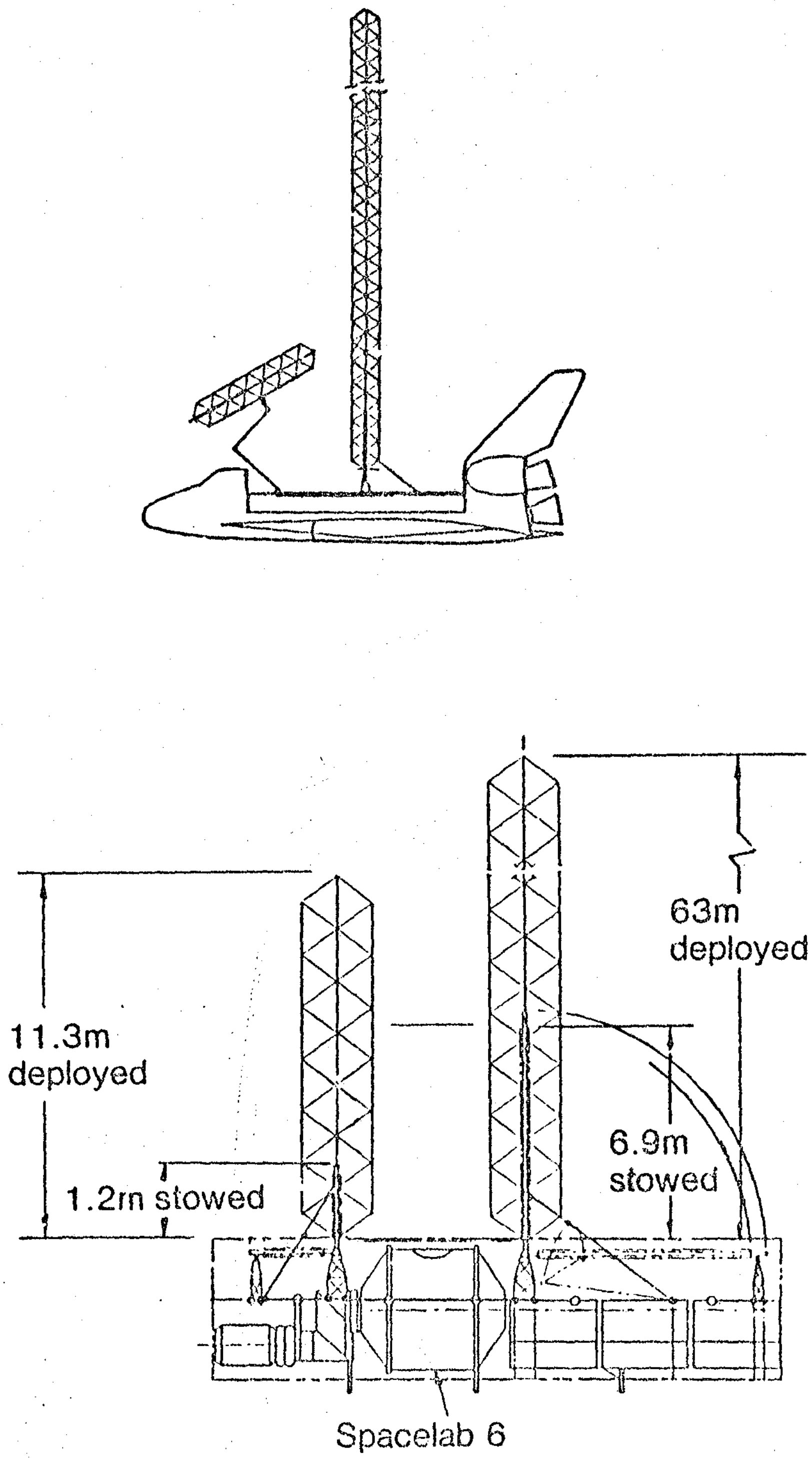
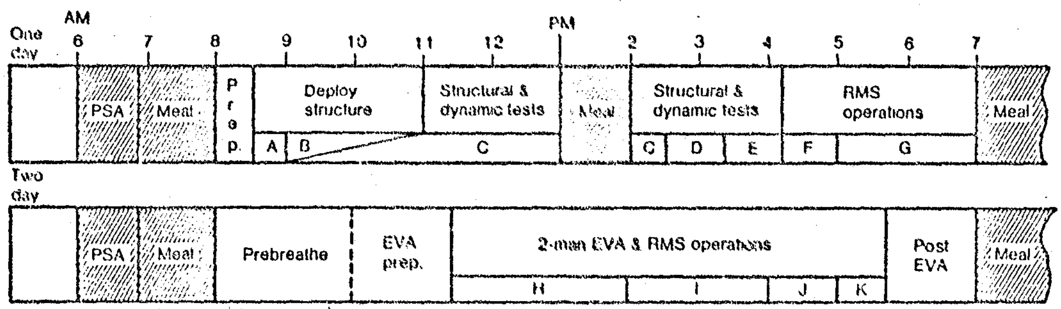


Figure 4-8. Experiment Concept 5

4.2.6 EXPERIMENT CONCEPT TIMELINES COMPARISON. The two-day operations sequence and timeline for concepts 1 and 2A are shown in Figure 4-9. The first experiment day is devoted to the structural and dynamic tests. The structure is rotated, the rails extended, and the structure is deployed. Dynamic tests are performed on the structure when it is partially deployed as well as when it is fully deployed. The tests determine the modes, damping effects, and the interaction of the structure with the Orbiter. In addition, dynamic tests will be performed on the RMS in various extended positions. The RMS will be exercised to inspect the deployed truss with the TV camera and to translate, position, and attach selected hardware components to the structure.



A — Rotate truss & extend rails
B — Deploy truss
C — Excite modes/test damping/effects on DAP
D — Nonlinear damping test
E — RMS arm extended effects
F — Truss inspection with RMS-TV
G — Practice exercises
H — Hardware instl/repair exercises
I — Removal & restowage
J — Retract truss
K — Stow truss

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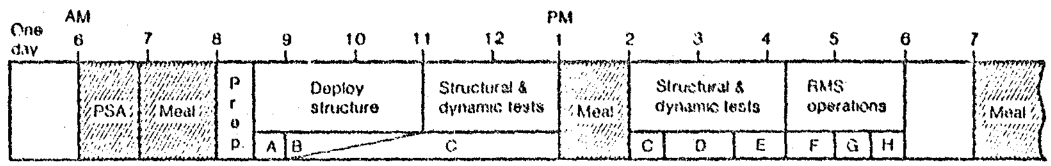
Figure 4-9. Operations Sequence and Timeline for Experiment Concepts 1 and 2A

During the second day the EVA/RMS operations are performed. The Pilot and Mission Specialist will perform hardware installations (subsystem modules, structural components, joints and couplings, cables and harnesses), undertake maintenance and repair exercises, and assist in the retraction and stowage of the truss. They will use the RMS to assist them in these operations.

The operations sequence and timelines for concepts 2, 3, and 4 are shown in Figure 4-10. The structural and dynamic tests are the main operations performed. The structure is rotated, the rails are extended, and then it is deployed. Dynamic tests are performed on the structure when it is partially deployed as well as when it is fully deployed. The tests determine the modes, damping effects, and the interaction of the structure and the Orbiter. In addition, dynamic tests will be performed on the RMS in various extended positions. The RMS will be exercised to inspect the deployed truss with the TV camera and to assist in the retraction and stowage of the truss.



A — Rotate truss & extend rails
B — Deploy truss
C — Excite modes/test damping/effect on DAP
D — Nonlinear damping test
E — RMS arm extended effects
F — Truss inspection with RMS/TV
G — Retract truss
H — Stow truss

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Figure 4-10. Operations Sequence and Timeline for Experiment Concepts 2,3, and 4

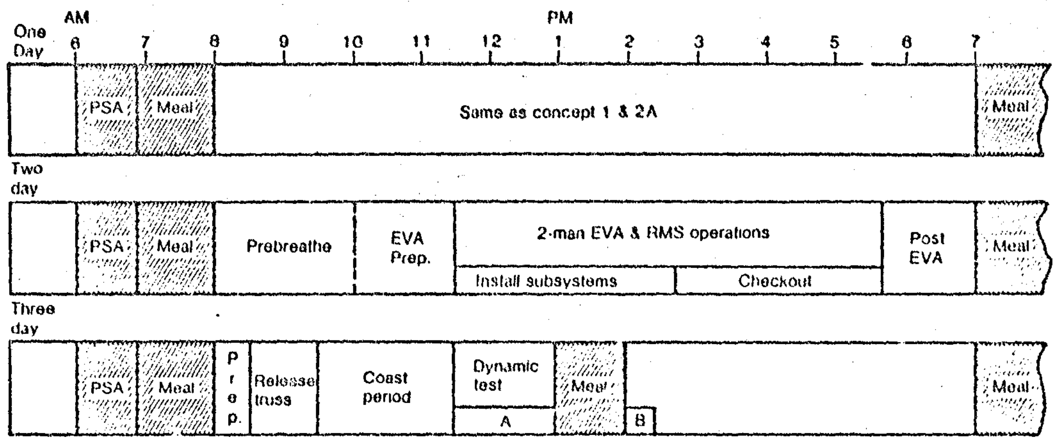
The concept 2B three-day operations sequence and timeline is shown in Figure 4-11. The first day's operation is identical to concepts 1 and 2A and covers mainly the structural and dynamic tests.

The second day entails EVA operations with RMS assist. The two astronauts will install and check out the additional subsystems; e.g., power package, telemetry package, RF control package, required to perform the free flying experiment.

The third day is devoted to the free-flying operation. The truss is separated from the Orbiter and placed in a gravity gradient position in proximity to the Orbiter by its RMS. The Orbiter flies in formation with the structure and receives the dynamic test information from the structure. After the tests are completed the structure is deboosted out of orbit.

The two-day operations sequence and timeline for concept 5 is shown in Figure 4-12. The first day's operation is identical to concepts 1 and 2A and covers mainly the structural and dynamic tests.

The second day entails the deployment of the second truss, its attachment to the first truss, and the retraction and stowage of both trusses. EVA to demonstrate its usefulness in assisting this operation is performed by two astronauts. In addition, the RMS is used to translate and position the second truss as well as to assist in the retraction and stowage of both trusses.



A — Excite modes/measure damping
B — Activate re-entry package

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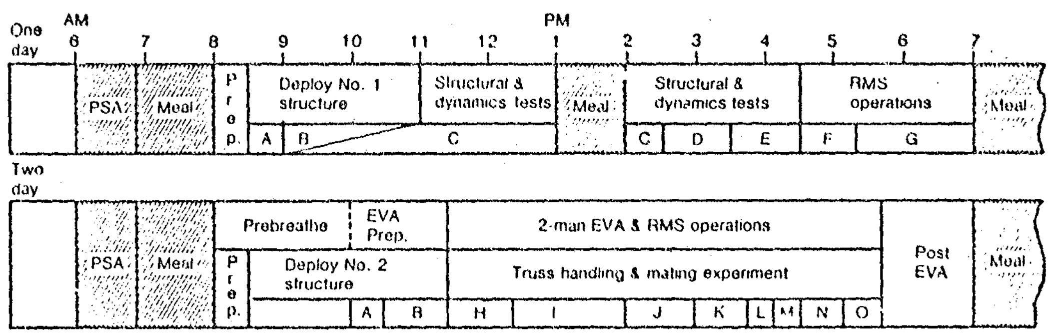
Figure 4-11. Operations Sequence and Timeline for Experiment Concept 2B



A — Rotate truss & extend rails
B — Deploy truss
C — Excite modes/test damping/effects on DAP
D — Nonlinear damping test
E — RMS arm extended effects
F — Truss inspection with RMS-TV
G — Practice exercises
H — Attach grapple & truss mating hardware
I — Separate & maneuver truss with RMS
J — Attach truss 2 to truss 1
K — Detach truss 2 & reattach to orbiter
L — Retract truss 2
M — Stow truss 2
N — Retract truss 1
O — Stow truss 1

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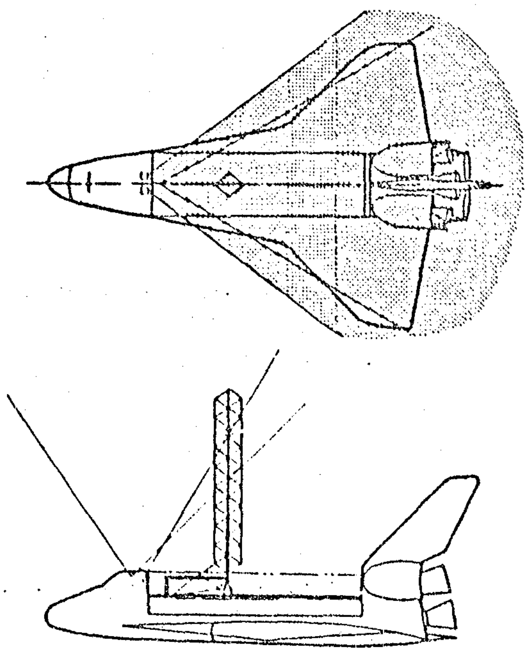
Figure 4-12. Operations Sequence and Timeline for Experiment Concept 5

4.2.7 ORBITER COMPATIBILITY EVALUATION. The viewing angles from the aft cabin windows were checked to determine the degree of visibility related to performing the desired experiments for each of the experiment concepts. Figure 4-13 illustrates the available viewing angles and the field of view (FOV) overlap from the overhead windows and the aft viewing windows. The rating for each of the concepts is as indicated. The concepts with the structure supported aft in the cargo bay have excellent viewing. The forward supported structures have acceptable viewing except for concept 4, where the structure is deployed horizontally with respect to the cargo bay. Concept 4 would make viewing for RMS operations very difficult. It also partially obstructs the radiation panels on the cargo bay doors.



Concept	FOV evaluation
1	Fair
2 & 2A	Good
3	Excellent
4	Poor
5	Excellent

Concept 2 shown

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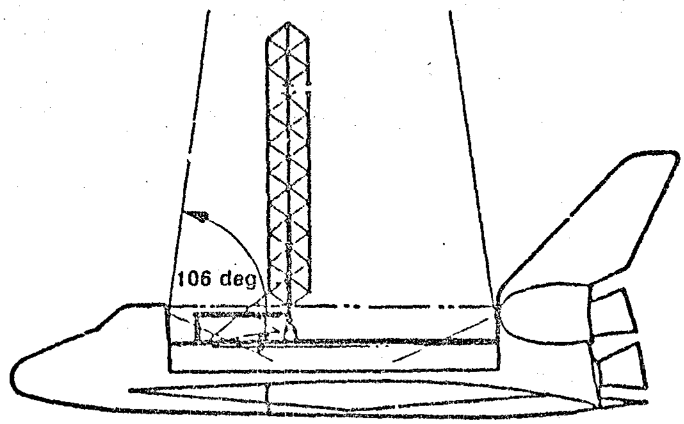
Figure 4-13. Aft Cabin Viewing Evaluation

The FOV from the Orbiter's closed circuit TV (CCTV) cameras covers the areas required to observe the experiments more than adequately. Relative ratings for CCTV viewing are given in Figure 4-14, which also shows the FOV from the forward and aft TV cameras.

The RMS working limits to perform experiments were evaluated as shown in Figure 4-15. This evaluation shows forward attachment of the SCE to be the best for RMS operations.



Concept	CCTV viewing eval
1	Good
2 & 2A	Excellent
3	Excellent
4	Fair
5	Excellent

Concept 2 shown

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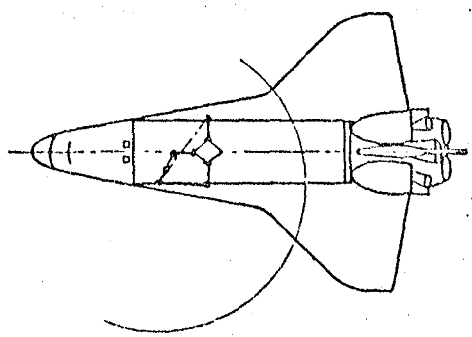
Figure 4-14. CCTV Viewing Evaluation

Concept	RMS working limit evaluation
1	Excellent
2	Excellent
3	Good
4	Good
5	Excellent

Concept 2 shown



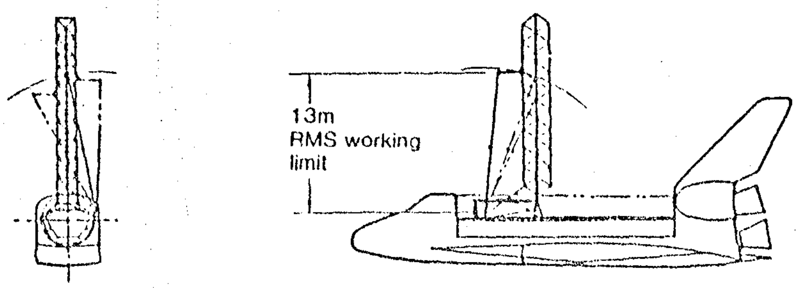


Figure 4-15. RMS Working Limits Evaluation

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4.2.8 EXPERIMENT CONTROL CONCEPTS. Several approaches are feasible for control, monitoring, and data collection for the SCE. One approach is to utilize the Payload Station Standard Switch Panel (SSP) for SCE power control, switch discretes, and discrete monitoring. The interface between the SSP and the SCE would be hardwire through the Payload Station Distribution Panel. In conjunction with the Payload Specialist operation of the SSP, the Mission Specialist at the Multifunction CRT Display System would initiate equivalent digital data and logic discretes with the GPC through the MDM-PF1. The Mission Specialist would monitor analog status functions. The SCE would require instrumentation, A/D conversions, and PCM encoding for data recording and CRT display input, and a D/A converter for the digitized analog, excitation signals. The GPC software would have to generate analog equivalent excitation signals, cyclic sequencing of mechanisms, and process display data.

The use of Convair-developed Digital Integrating System (DIS) processors would reduce the operational complexity of two-man (Payload Specialist and Mission Specialist) experiment operation to a Payload Specialist operation. All software (other than for data recording) would be resident in DIS, eliminating a complex software interface. Control and monitor functions would be over the DISMUX bus, eliminating hardwire links between the SCE and Payload Specialist Station, other than for power, and safety hardwire control of the SCE jettison. The DIS employs a Z8002 microprocessor, 1K RAM, and 4K PROM chips. This concept is shown in the Figure 4-16 block diagram.

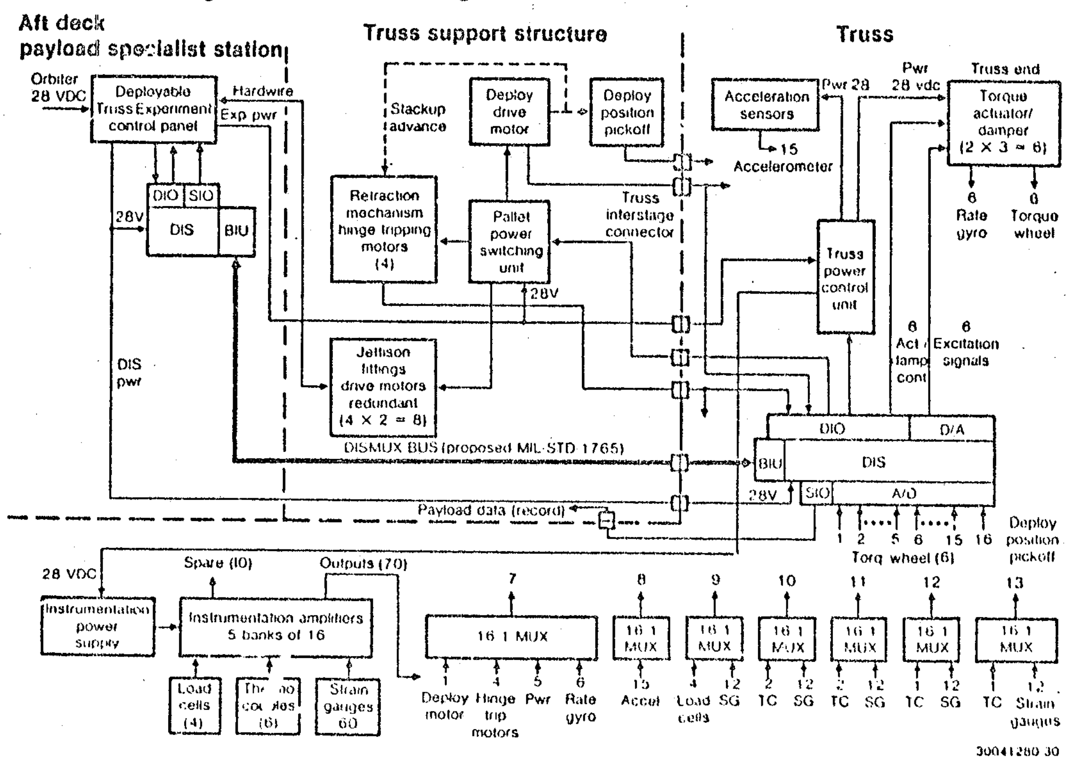

Figure 4-16. DISMUX Control Concept Block Diagram

The SCE Control Panel contains switches, indicators, fuses, and a digital readout. The Discrete Input-Output (DIO) module accepts discrete commands and provides a discrete status info to the SCE Control Panel. The Serial Input-Output (SIO) module provides the data to the digital readout. The DIS associated with the SCE Control Panel has all the software associated with control and monitor functions. The Bus Interface Unit (BIU) module communicates with the truss-mounted DIS over the DISMUX bus. The truss DIS transmits motor control discretes to the TSS Power Switching Unit, and power control discretes to the Truss Power Control Unit via a DIO module. Monitor discretes from mechanism limit switches and motor control relays are fed back to the DIO. The DIO operates the Torque Actuator/Dampers in the actuate or dampling mode. Various analog excitation signals are available under software control while in the actuate mode. The bulk of the truss DIS software is allocated to control of instrumentation multiplexing and instrumentation data processing. The analog data from the rate gyros, motor current monitors, accelerometers, strain gauges, thermocouples, load cells, and power monitors are multiplexed into a 16-channel A/D converter module along with torque wheel position and deployment position. The DIS packs the data format and transmits it through the SIO module to the Payload Data Interleaver (PDI) input. Few caution and warning functions are provided to the Payload Specialist and Mission Specialist. One is the status of the SCE jettison latches, and another is any redline power dissipation condition. The DIS units could be supplanted with the more sophisticated NASA Standard S/C computers at an anticipated higher procurement cost.

Another SCE control concept is to have single Payload Specialist operation and self-contained software, but replace the SCE Control Panel to SCE control bus with hardwire functions, eliminating the truss-mounted processor. This concept is shown in Figure 4-17. An adaptation of the Centaur-In-Shuttle (CIS) Control Unit with a Z80, 8-bit microprocessor, would be utilized. The CIS Control Unit has relay outputs that can directly operate the mechanism motors, and switch system supply power. Analog data from the deployment position and rate gyros is A/D converted for status and control purposes in the Control Unit. The software controlled excitation signals are hardwired from the Control Unit D/A converter module. Various parameters (including motor current monitor) are formatted in the Control Unit for transmittal through the SIO to the PCM Encoder for interleaving into the instrumentation data stream to the PDI. The PCM Encoder performs multiplex control, A/D conversion, data formatting, and data stream transmittal.

The hardwire concept is lower cost than the DIS concept because the CIS Control Unit is less complex than DIS, and the PCM Encoder is simpler than the truss-mounted DIS. The DISMUX bus is replaced with approximately 30 hardwire functions.

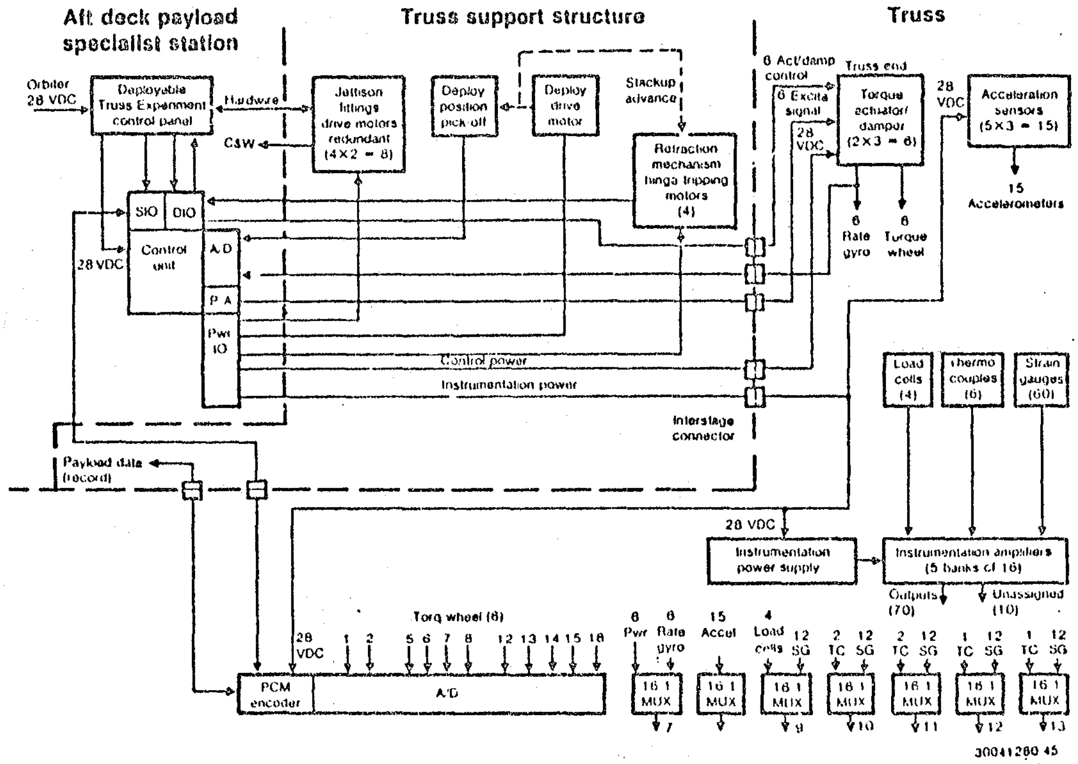

Figure 4-17. Hardwire Control Concept Block Diagram

4.2.9 PRELIMINARY COST COMPARISONS. Very preliminary rough order of magnitude (ROM) costs were estimated for several versions of the SCE for use in comparison of the various alternatives. These costs were generated parametrically based on the very preliminary hardware definitions. The absolute cost values require more detailed hardware definition and additional cost analysis for improved confidence; however, the relative cost values shown in Table 4-1 are representative for each of the concepts. Costs are shown in millions of 1981 constant dollars, and prime contractor fee is excluded. Estimates were made for nonrecurring (development) costs and the production of the flight unit. Only one unit is produced and all ground system level testing is accomplished with the flight unit, which is then refurbished for the actual flight test.

As noted, triangular beam concepts were also estimated in lieu of diamond beams, reducing the overall cost 9 to 22%, depending on the specific experiment.

Shuttle user charges that vary according to the experiment accommodation mode are not included. In all cases the user charge load factor would have been based on experiment length in the Orbiter payload bay. However, user charges are not considered for NASA payloads.

Table 4-1. Alternative Concept Preliminary ROM Cost Estimates (1981 $M)

Concept	Development cost	Unit cost	Total
1	5.5	1.2	6.7
1-1*	5.1	1.0	6.1
2	4.5	1.4	5.9
2A	6.5	1.6	8.1
2B	11.2	2.7	13.9
3	4.8	1.5	6.3
4	3.2	0.8	4.0
5	5.5	1.7	7.2

*Triangular vs diamond beam reduces cost by an average of 15%

4.2.10 CONCEPT SELECTION. The numerical evaluation of the candidate experiments is presented in Table 4-2. The evaluation criteria are identified and the relative merit of each concept against these criteria is rated. The evaluation criteria were chosen to screen the concepts against the major performance capabilities and program and operational concerns. The rating system is one to five, with five being the best rating for each criterion. No relative weighting factors were applied to the selection criteria.

The sum of the rating factors indicates highest potential benefits for concepts 1 and 2A. However, concepts 2 and 4 are lowest in cost. Concepts 3, 4, and 5 have the lowest potential benefits, which makes them the least desirable approaches. Concept 2B exceeds the $10M program cost guideline, which narrows the choice to concepts 1 and 2A.

By dividing the estimated prog im costs into the sum of the rating factors, it is seen that concept 1 has the highest benefit/cost ratio, with concept 2A a close second.

The final choice between concepts 1 and 2A was made on the basis of the greatest potential benefits to be derived from concept 2A. Concept 2A provides a longer truss structure and has the potential for even greater length depending upon the final payload envelope allowed. Concept 1 has no growth capability and has marginal length for performing structures and dynamics testing. Concept 2A was selected because of its superior overall capabilities and high cost effectiveness ratio.

Table 4-2. Experiments Concepts Numerical Evaluation
(Scale of 1-5, High Numbers Best)

Evaluation Criteria	Experiment Concepts						
	1	2	2A	2B	3	4	5
Performance capabilities							
• DAP effects testing	3	5	5	5	5	2	5
• Structural dynamics testing	3	4	4	5	5	2	5
• RMS operations	5	2	5	5	2	1	3
• EVA operations	5	0	5	5	0	0	3
• Deployment/retraction	4	4	4	3	4	2	5
• Suitcase experiments	5	0	5	3	0	0	2
Program & operational							
• More flight opportunities	5	4	4	4	1	5	1
• Orbiter compatibility	3	4	4	4	4	2	5
• Potential for multi-mission options	3	4	5	2	4	1	4
• Minimal development risk	4	4	4	2	4	5	4
$\sum$ Rating factors	40	31	45	38	29	20	37
Benefits/$M cost* ratio	5.97	5.25	5 56	2.73	4.60	5.00	5.14

*Shuttle user charges not included

4.3 PRELIMINARY DESIGN

The preceding trades and concept evaluations resulted in the selection of a deployable tetrahedral truss supported in the Orbiter by a support structure. The baseline configuration assumes an arrangement whereby the basic experiment shares space in the forward section of the payload bay with an MPS experimental pallet on a flight accompanied by deployed satellite payloads. Additional suitcase experiments for EVA/RMS experiments are integrated into the SCE payload.

4.3.1 STRUCTURAL AND MECHANICAL DESIGN. The general arrangement for the basic SCE is shown in Figure 4-18. The 50.1m truss assembly is stowed in its deployment rail in a short, flat packaging envelope. The initial deployment sequence is performed using the RMS to accomplish the following unlatch and rotation functions:

a. Unlatch forward latch

b. Rotate truss package 90 degrees

c. Deploy forward side members with lift and holddown arm

d. Deploy extension rail No. 1

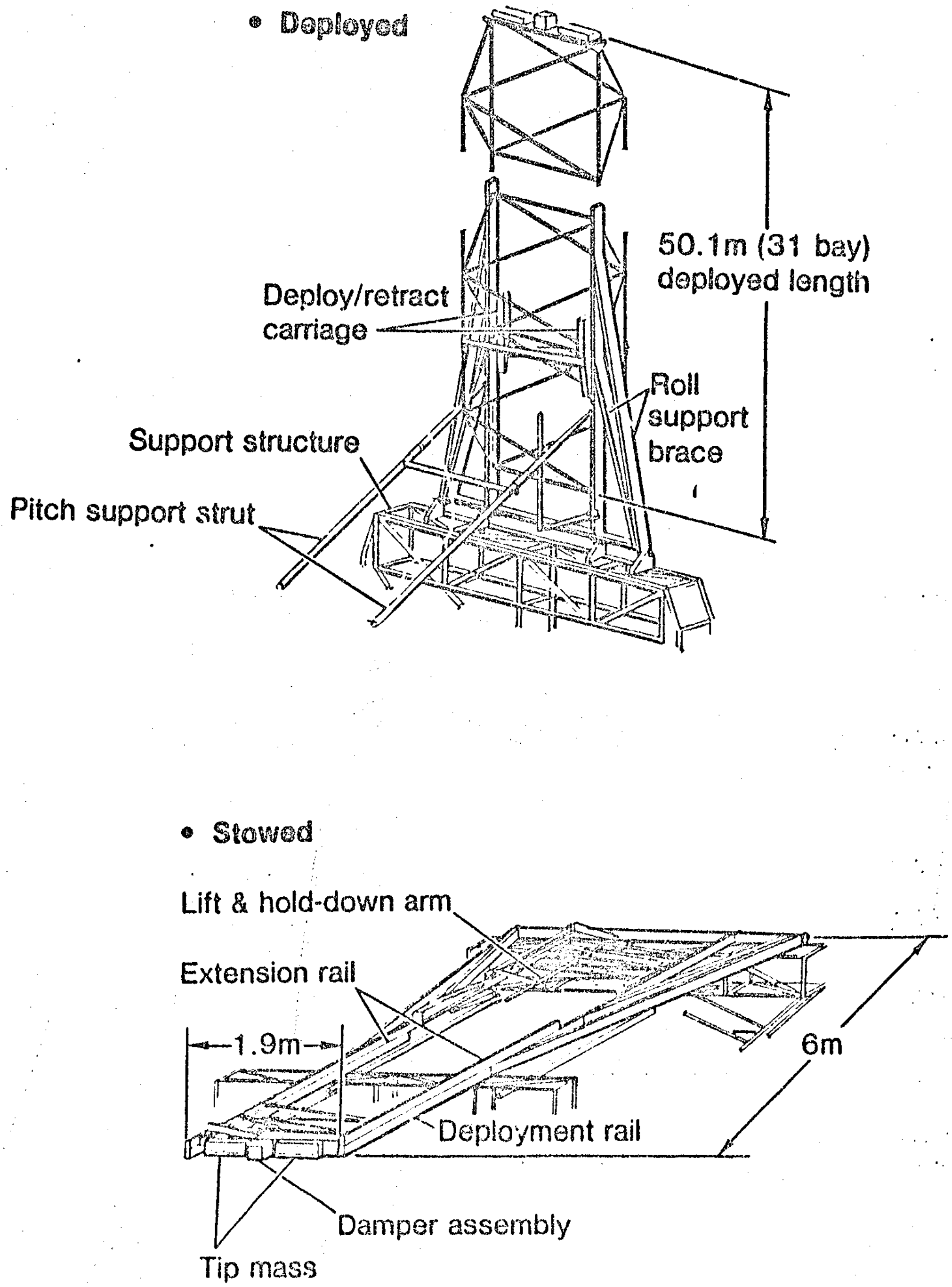

Figure 4-18. Basic Experiment General Arrangement

e. Deploy aft side members with lift and holddown arm

f. Deploy extension rail No. 2

g. Deploy tripper support arms

Following initial deployment, two motor-driven deploy/retract carriages sequentially deploy the tetrahedral truss - one bay at a time. The deployment rails provide moment reaction support of the truss structure at all times as each bay is deployed and latched open by self-locking hinges. This sequence may be stopped or reversed at any stage of deployment. For retraction the carriages act to automatically unlock or trip the hinges in a truss bay and collapse to a folded position.

The deployment rails are braced, as shown, to react pitch and roll moments for worst-case contingency loads that could occur during test, to ensure the safety of the crew and Orbiter. The pitch braces also serve as handholds for EVA in the vicinity of the support structure.

Characteristics and performance of the basic experiment are summarized in Table 4-3. Deployment/retraction rate was selected to limit power requirements to 0.5 kW. This is also the estimated power required to operate the dampers at the maximum damping ratio.

Table 4-3. SCE Performance and Characteristics Summary

Item	Value
Deployment/Retraction Rate	3 bays/min
Deploy/Retract Drive Speed	0.3m/sec
Power (peak)	500W
Damping Ratio (active)	0, 1%, 2%
Tip Mass	400 kg

The experiment deployable truss is to be stowed above the MPS payload in the forward section of the cargo bay within the space available. The truss and deployment rail is supported by a truss network support structure that transfers loads from the rail to the Orbiter structure. The support structure and its installation details are shown in Figure 4-19.

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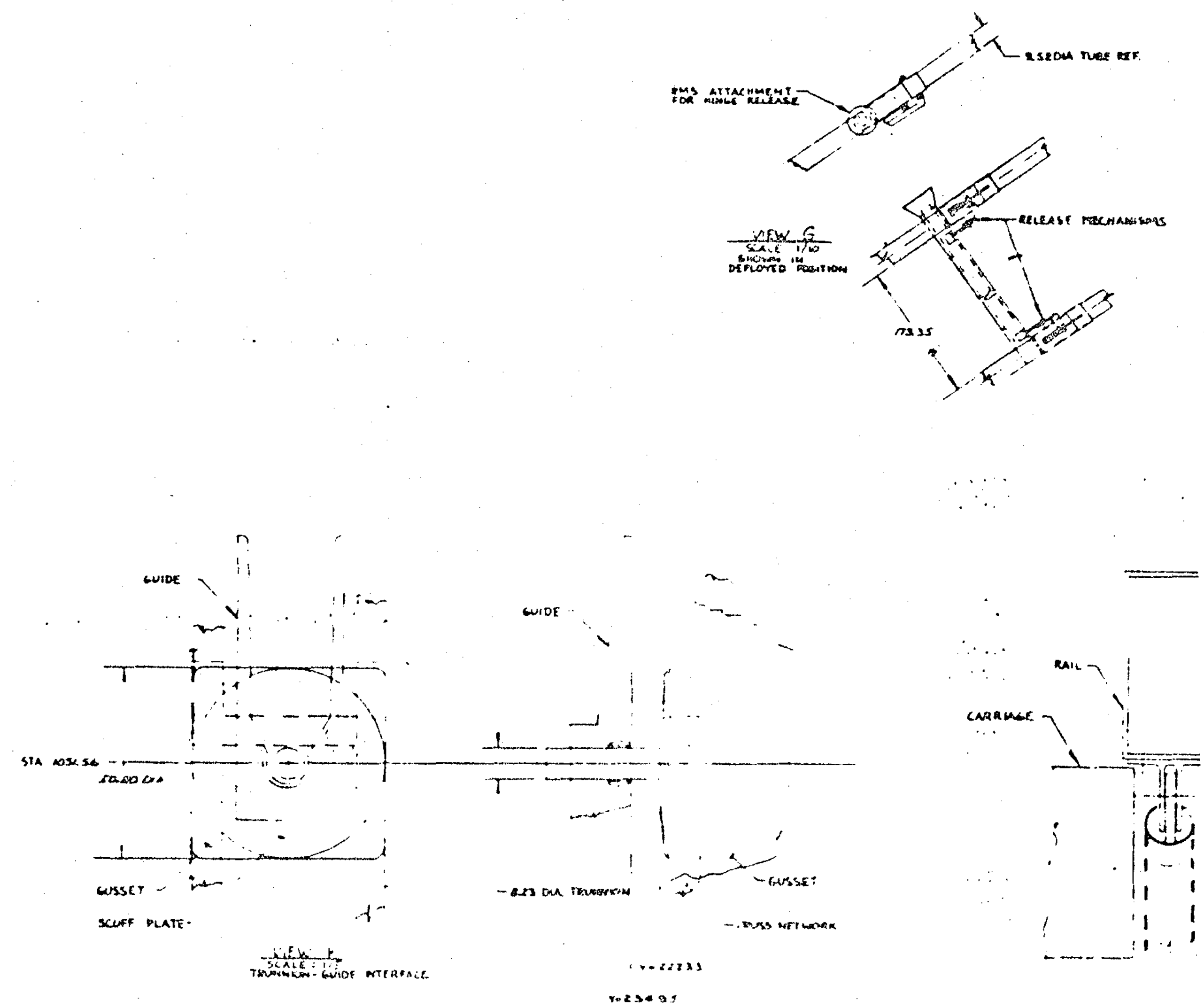

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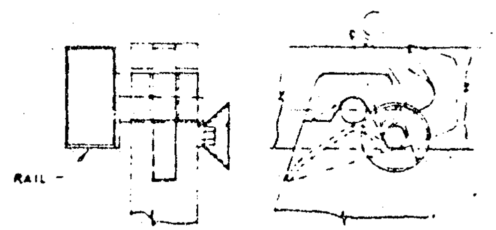

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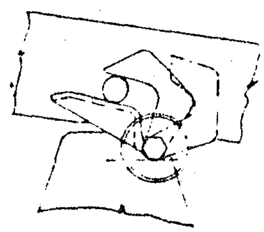

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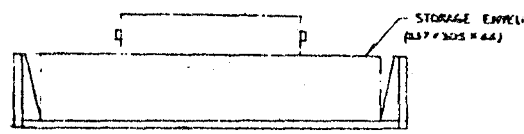

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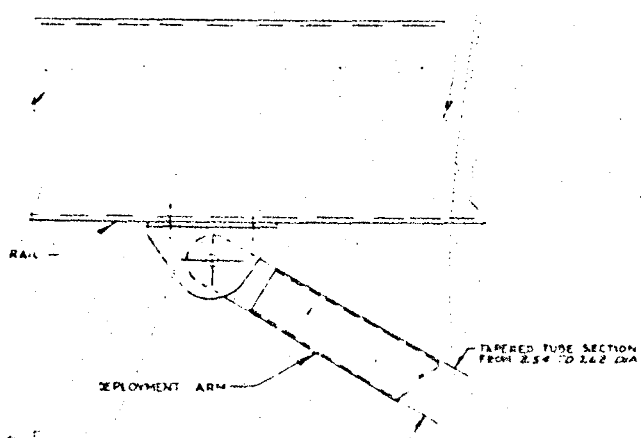

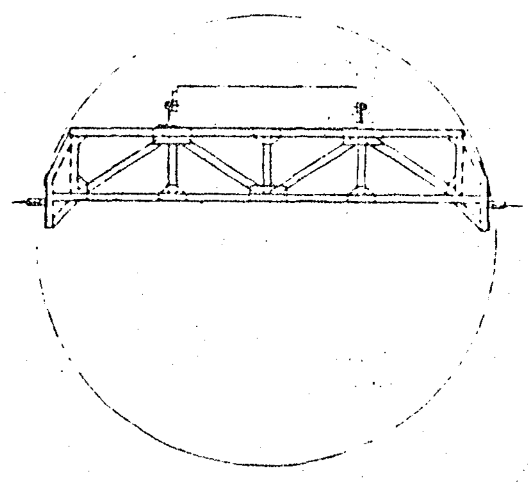

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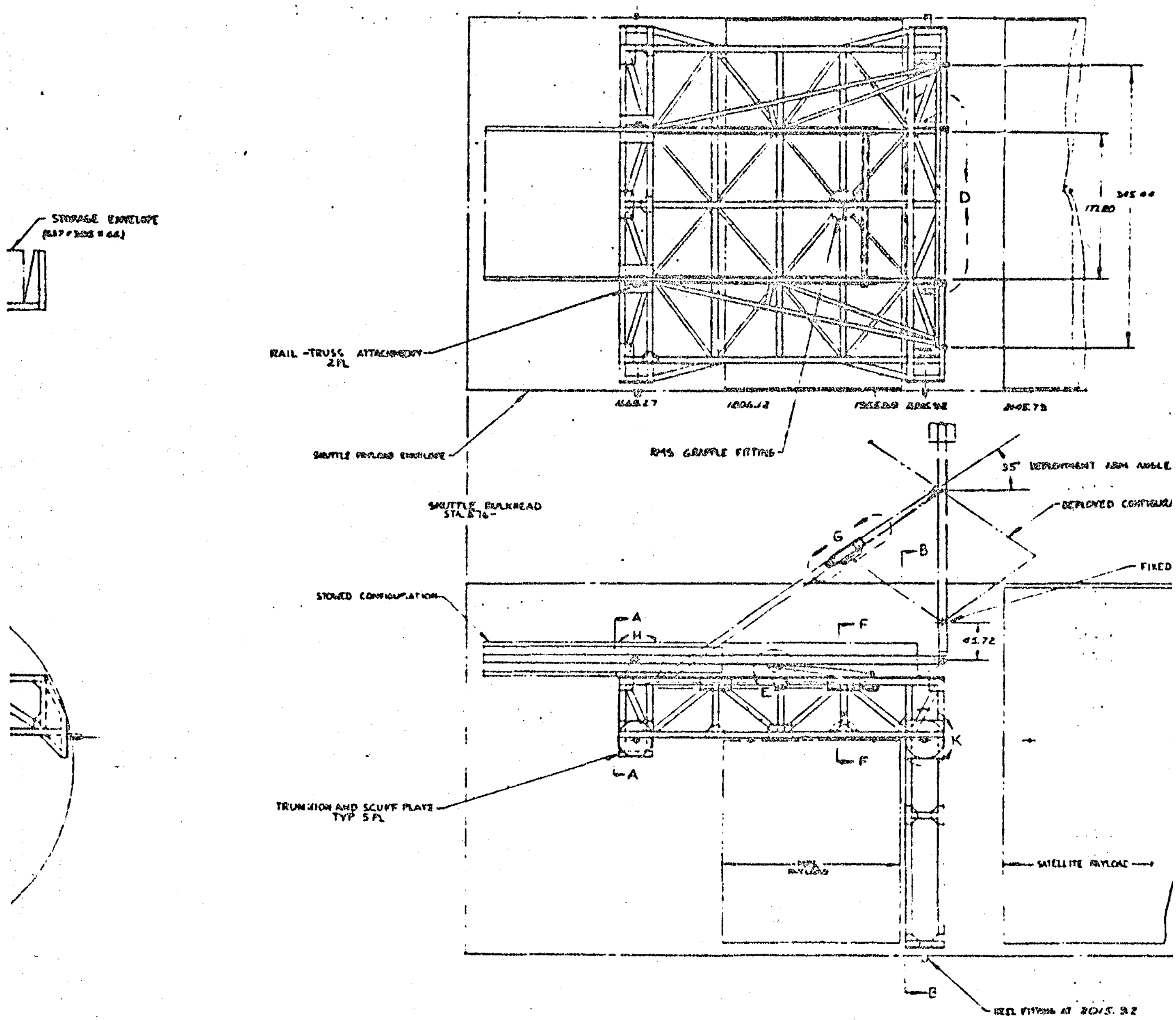

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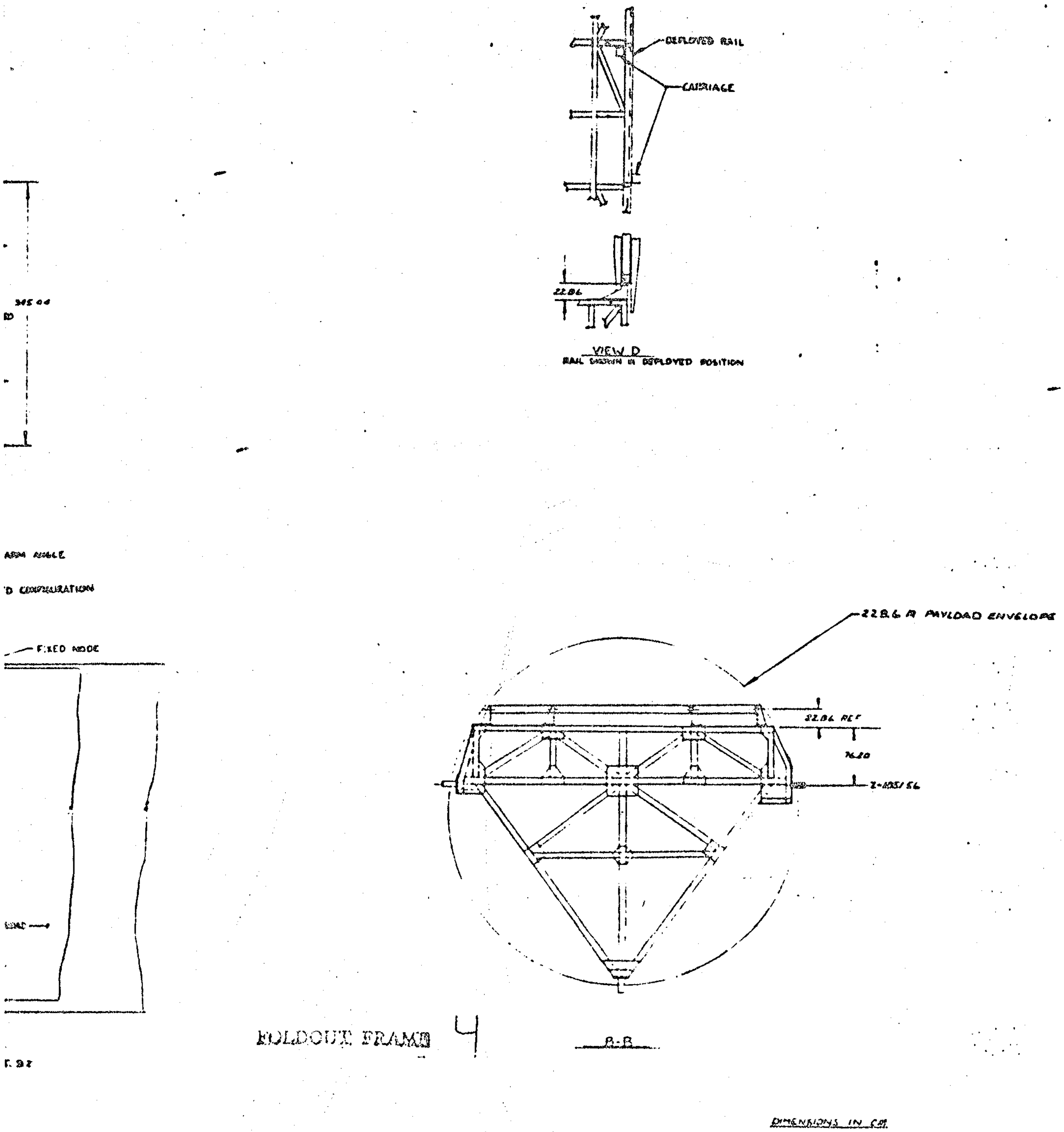

Figure 4-19. Experiment Support Structure and Installation

The support truss network is mounted in the Orbiter at available keel and longeron support stations. Since it is a requirement that the experiment have jettison capability, the keel and longeron fittings are of the active-deployable type. Jettison of the truss network is accomplished by operating the active Shuttle fittings. Removal of the entire truss from the Shuttle is performed with the RMS by engaging a grapple fitting located on the support structure, as shown. The support truss network provides a stowage space and support for suitcase experiments.

The support structure is located to provide adequate room for astronaut access into and out of the forward cabin. The support structure is constructed of 7.6cm square, thin-walled aluminum tubes with appropriate gusseting. Trunnions and scuff plates are attached at four places for longeron fittings. One keel fitting is provided. Each longeron fitting includes a standard guide-rail mechanism to facilitate deployment of the structure.

Six attachment fittings are provided to hold the deployment rail to the truss network. Four aft hinges allow the rail to rotate for deployment. Two forward latching supports hold the deployment rail in its stowed position. These two latch mechanisms interface with the RMS to release the rail for deployment and recapture the rail for stowage.

Two 9.5cm diameter, thin-walled aluminum struts provide pitch moment support needed at the base when the truss is deployed. The struts use "over-center" locking hinges. The two tubes are connected together by a crossbar that has an RMS connector to release the over-center hinges for restowage of the deployable truss assembly.

The FSS cradle B and the MPE support structure were evaluated for use in the experiment. Neither was selected. Instead, the single support structure for the deployable truss and the suitcase experiments and accessories stowage was selected.

The main criteria used to select the support structure were as follows:

a. Availability of active longeron and keel fittings with existing payloads on the baseline mission.

b. Location of stowed deployment rail to be above the MPS payload.

c. Maintaining the 48-inch, astronaut-access area aft of the aft cabin bulkhead.

d. Location of equipment stowage space between the stowed deployment rail and the MPS. With this configuration, it offers easy access to the equipment by the RMS and serves as a working platform for the astronauts.

Standard FSS cradles A, B, and A' (Reference 10) were considered with the four main criteria above. Cradles A and A' offered restricted inside geometry. They would require specially-designed supports between the cradle and the deployment rail. Cradle B is 54 inches long. This is too long to be used without shifting around existing payloads.

The Teledyne MPE support structure (Reference 9) was considered but didn't provide adequate height (in Z direction) to attach the deployment rail and clear the MPS payload. Use of the Teledyne structure would require extensive modifications to adapt it to the experiment.

The experiment preliminary design employs special bellmouth fittings for insertion of a hexagonal head drive so all of the initial deployment and final stowage functions can be performed either manually or with the RMS. The concept shown in Figure 4-20 employs the Universal Service Tool (UST) developed by Spar Aerospace Ltd. for NASA/GSFC. The UST is a motor-driven torque wrench which can use special torque tool attachments as driving fasteners. The UST attaches to the RMS standard end effector and is powered through an electrical interface. This device can also be used to install equipment modules on the basic experiment. An alternative approach would be a simple socket wrench mounted on a standard grapple fitting to use the wrist roll motion of the RMS to rotate the deployment drive shafts.

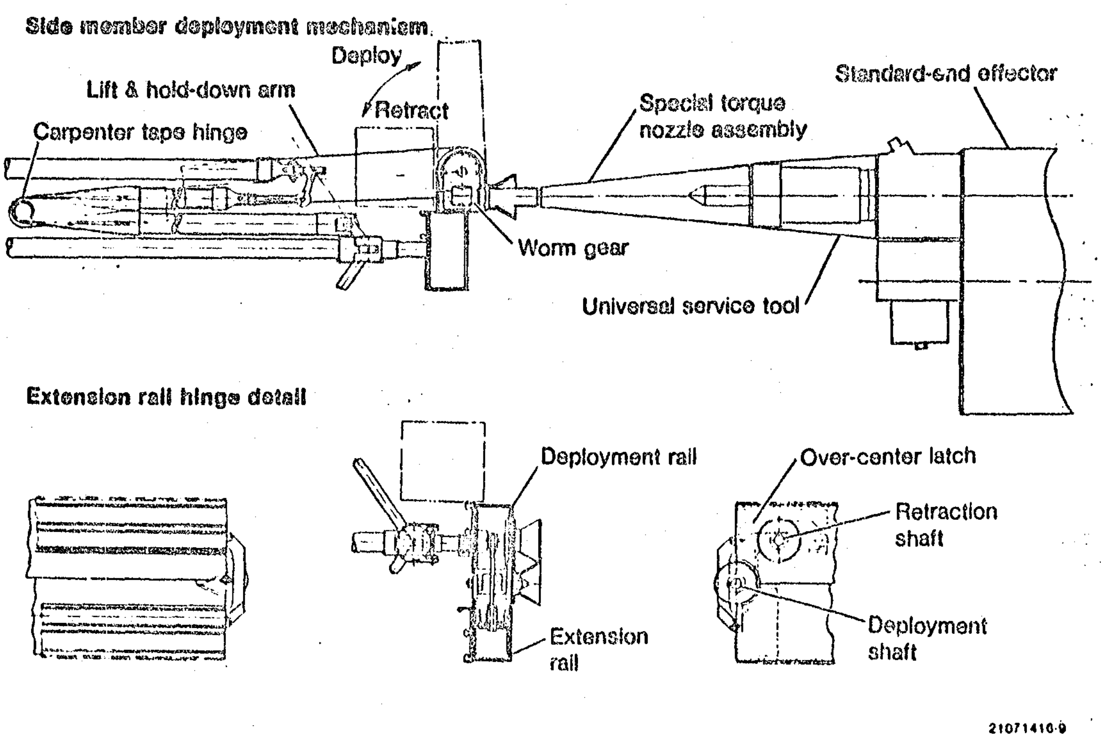

Figure 4-20. RMS Driven Deployment Mechanisms

Two truss deployment and retraction carriages deploy the tetrahedral truss and retract it for stowage. Figure 4-21 shows carriage details. The carriage translates along the deployment rail on a set of rollers in a track. Each carriage is driven by a brushless dc gear motor with a rack and pinion drive. A drive latch engages a node and roller fitting on the truss. The drive latch is disengaged by a rotary solenoid actuator after each carriage stroke.

During deployment, each drive latch engages a truss node fitting on opposite sides of the rail. The carriages drive in the deploy direction until a truss bay is open and locked. The drive latches are disengaged and the carriages return to pick up the next bay node fittings. Power and control signals are transmitted to each carriage by wind-up harnesses on a reel.

During retraction the drive latches remain disengaged as the carriages drive past the node fittings a short distance. On the retract stroke, the two hinges' trippers on each carriage unlatch the hinges in one bay so that, as the drive latches engage the node fittings, the bay will collapse and be retracted by the action of the carriages.

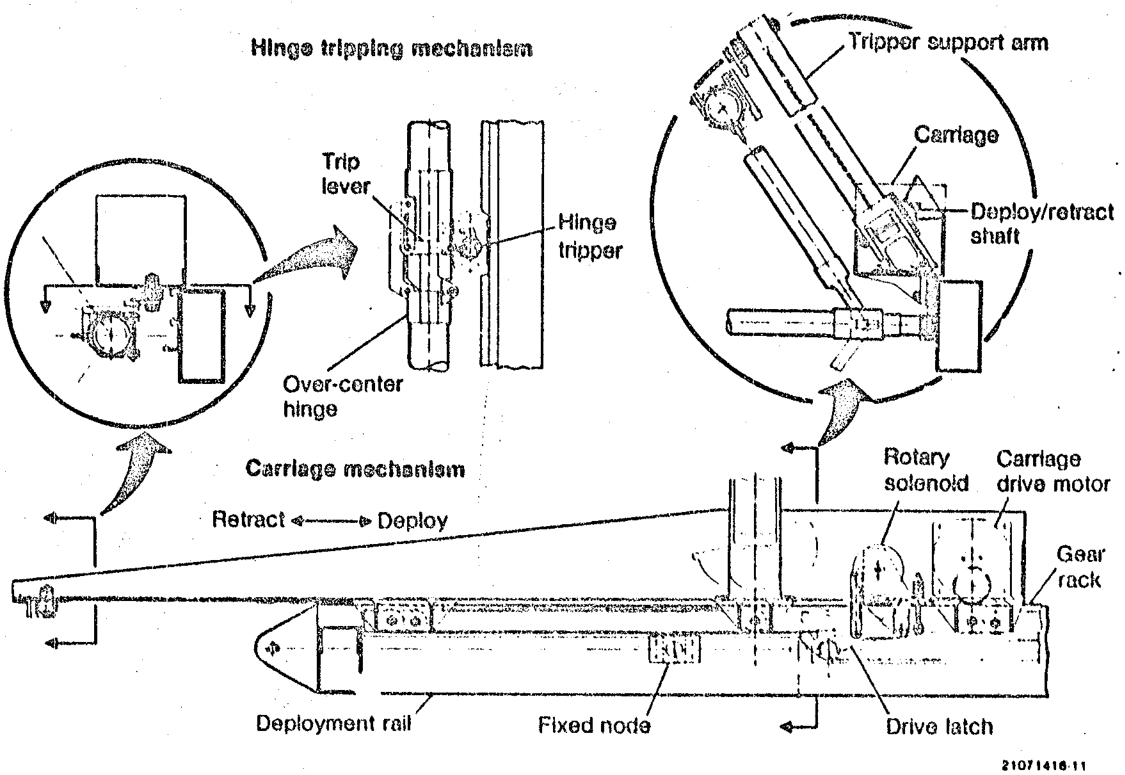

Figure 4-21. Deployment/Retraction Carriage Mechanisms

4.3.2 CONTROL, POWER, AND DATA ACQUISITION. The selected approach for the SCE control shown in Figure 4-22 uses a microprocessor controller: the Shuttle-qualified CIS Control Unit with a Z80, 8-bit processor. For instrumentation, a standard off-the-shelf PCM Encoder provides a 16 Kbps data stream to the Orbiter payload data interleaver for recording purposes. Hardwire interfaces are utilized between the deployable truss and support structure, and the Payload Specialist Station. Hardwire requirements are estimated in Table 4-4.

Revisions to the preliminary hardwire control concept include use of two carriage-driven drive latches with solenoid actuation for hinge tripping instead of four hinge tripping motors, two brushless motors instead of one conventional dc carriage-drive motor, two digital encoders replacing one analog pickoff, addition of another jettison fitting dual motor, and addition of redundant rate gyro data slots in the PCM data stream to complement the torque wheel data redundancy. The actual distribution of multiplexer inputs will be more complex than illustrated to prevent loss of all of any one type of data source because of the malfunction of a single 16:1 multiplexer. The Control Unit outputs a 4 Kbps data stream, which the PCM Encoder interleaves with the multiplexed data sources.

The relay switches for the conventional motor carriage drive have been replaced with low-power-dissipation hybrid switches, currently under company development, for the brushless motor drive. Both types of switches mount on the same form factor module cards in the Control Unit.

The selected concept provides flexibility via software control, has digital accuracy, utilizes reliable drive electromechanisms, and promises moderate cost with only two qualified electronics units (Control Unit and PCM Encoder) plus a control panel, in addition to the deployable truss electronics and electromechanisms.

A fully digital control mechanization is employed for the carriage drive control concept of Figure 4-23. A brushless Sm-Co motor with oversaturation resistance, high thermal conductivity, and reliable life is employed. An incremental optical encoder mounted directly on the motor shaft serves three purposes: (1) provides pulse rate feedback for a tight velocity loop around the motor, (2) provides incremental position for fine-position tracking of the two carriages and for carriage position sequencing, and (3) provides the rotor position for electronic commutation.

An 8 parallel bit output, absolute optical encoder completes slightly less than one full revolution for maximum carriage travel. This eliminates the need for the Control Unit to resolve any position ambiguity, and prevents loss of control recovery in case of a carriage position data transient. Using optical encoders that inherently provide high digital accuracy, the speed reduction backlash is the only position error source. A typical 30 arc-minute maximum backlash is an equivalent 0.018cm carriage travel, which is negligible compared to the allowable 0.3cm maximum carriage position difference.

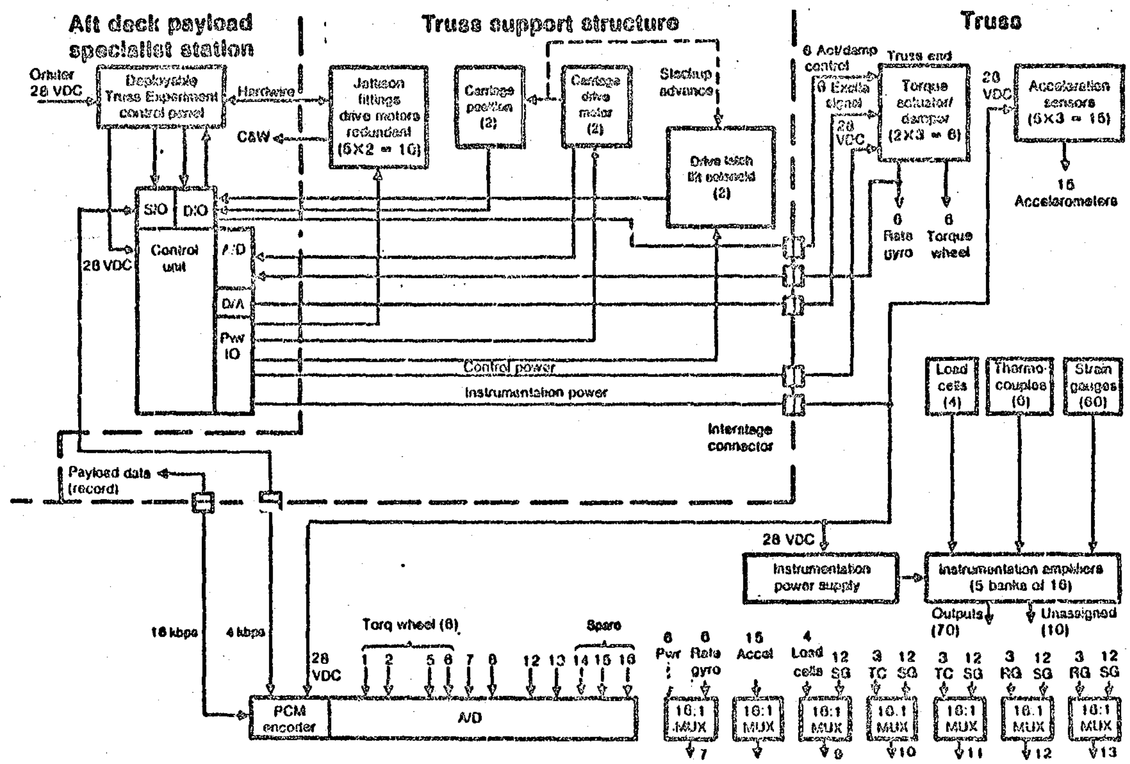

Figure 4-22. Selected SCE Hardwire Control Concept

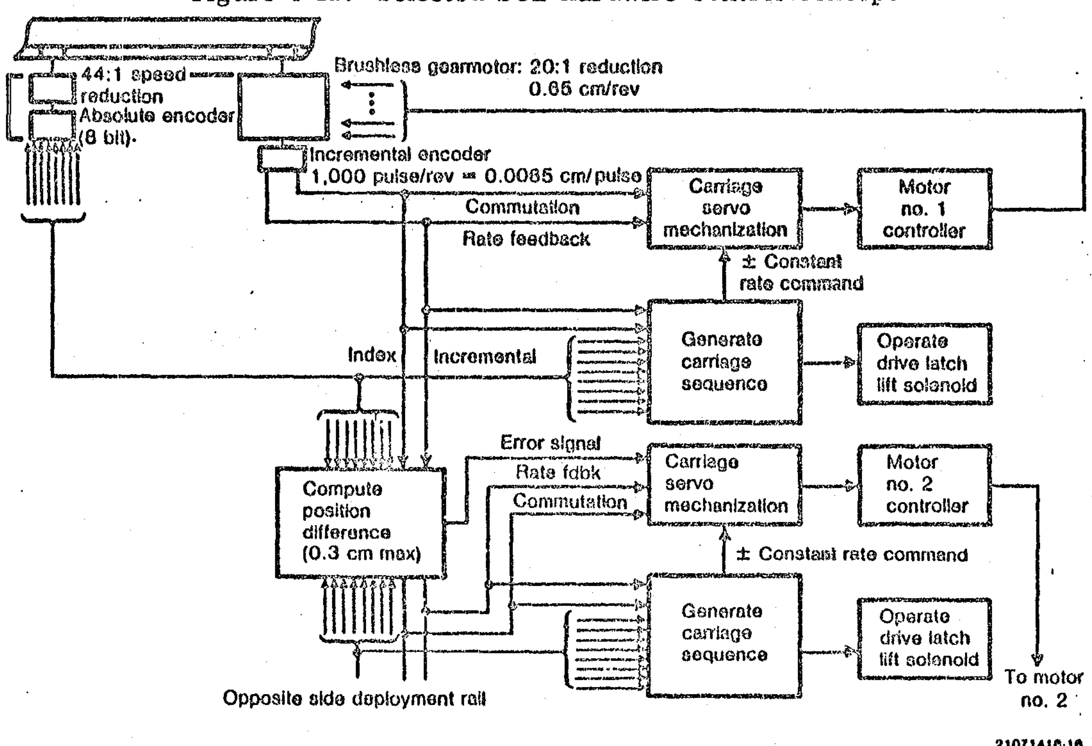

Figure 4-23. Carriage Drive Control Concept

Table 4-4. SCE to Orbiter Interface

Function		Number of Wires
• DEPLOYMENT		
- Deploy Motor Drive 0.5 kg-m @ 10 A		33 x 2 = 66
- Motor Position Feedback	Logic Level	3 x 2 = 6
- Carriage Position Feedback	Logic Level	9 x 2 = 18
- Encoder Power 5VDC 14 A		2 x 2 = 4
- Drive Latch Lift Solenoid Drive 0.2 A		2 x 2 = 4
- Drive Latch Position	Logic Level	4 x 2 = 8
- Motor Temperature Sensor	Low Level	2 x 2 = 4
• JETTISON		
- Active Fitting Motor Drive	0.1 A ac	10 x 2 = 20
- Active Fitting Latch Position	Logic Level	2 x 5 = 10
- Truss Support Structure		
- Separation Sensors	Logic Level	2 x 4 = 8
• CONTROL		
- Switched Truss Control Power	10 A	2
- Actuate/Damp Selection	Logic Level	7
- Actuate Excitation	Low Level	7
- Rate Gyro Output	Low Level	6 x 2 = 12
- Torque Motor Temp Sensor	Low Level	6 x 2 = 12
• INSTRUMENTATION		
- Switched Instrumentation Power 3 A		2
- Control Unit Data Bus	Logic Level	2
- PCM Data Bus	Logic Level	2
	TOTAL	194

In the Figure 4-23 diagram, the top carriage is shown as a self-contained loop with position information being used to generate the commanded carriage traverse sequence at constant rate (plus appropriate ramp/de-ramp velocities), with the carriage servo using pulse rate feedback for computing the rate error control signal. The opposite side carriage servo, in addition, tracks the other carriage by computing a position difference error control signal that modifies its operating rate to minimize the position difference. The Control Unit has automatic torque limiting software in case a temporary electrical power dropout scrambles the control loop. This prevents any equipment damage, and permits the Payload Specialist to reestablish the sequence for deployment or retraction. A typical sequencing of the carriages is shown in Figure 4-24.

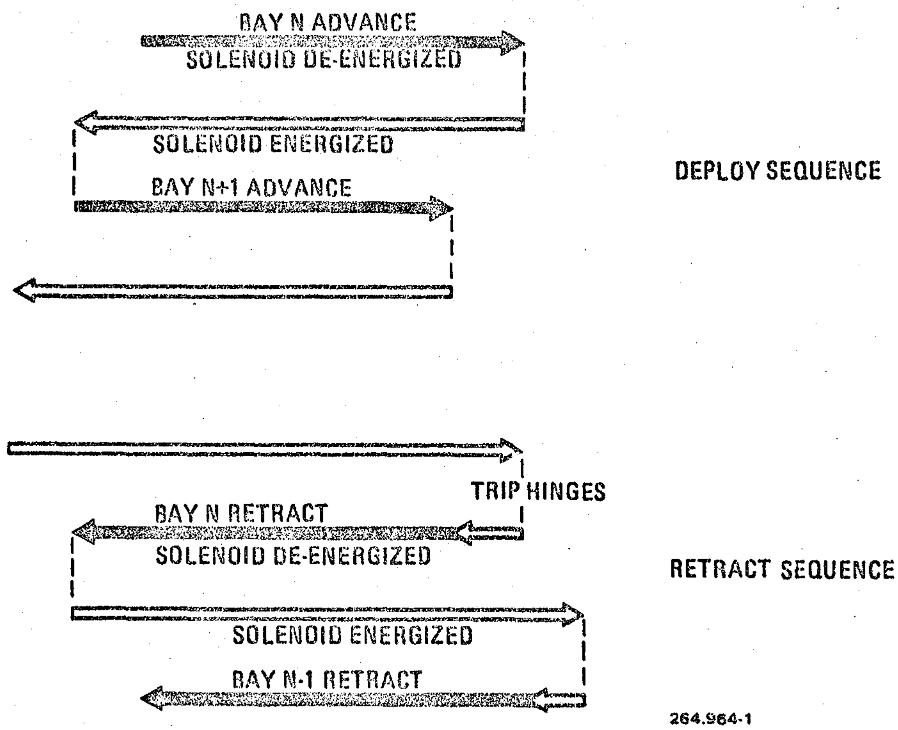

Figure 4-24. Carriage Sequence Diagram

The carriage servo control loop mechanization illustrated in Figure 4-25 is contained in the Control Unit processor software and appropriate input/output. The carriage sequence software generates the commanded constant rate with reamp/de-amp rates, and the solenoid driver activate/deenergize signal in accordance with the desired deployment or retraction sequence as a function of carriage position.

The rate stabilization software differences the commanded rate and the pulse rate feedback for a rate error signal. Control filtering is provided for high stability margin servo performance. The opposite side carriage servo also sums in a position error signal with appropriate proportional plus integral compensation for good position tracking performance.

The pulse width modulation software generates a pulse width modulated output as a function of the control error signal, to obtain a low power dissipation mode of motor operation. The stator field winding switches are sequenced by the commutation sequence software for correct winding energization by the pulse width modulated signal. The pulse width modulation software includes a time-out function; therefore, a too long, high modulation signal (representing long duration high torque) results in automatic shutdown with Payload Specialist notification. The torque switches use HEXFET devices with high gain, and a positive temperature coefficient without secondary breakdown.

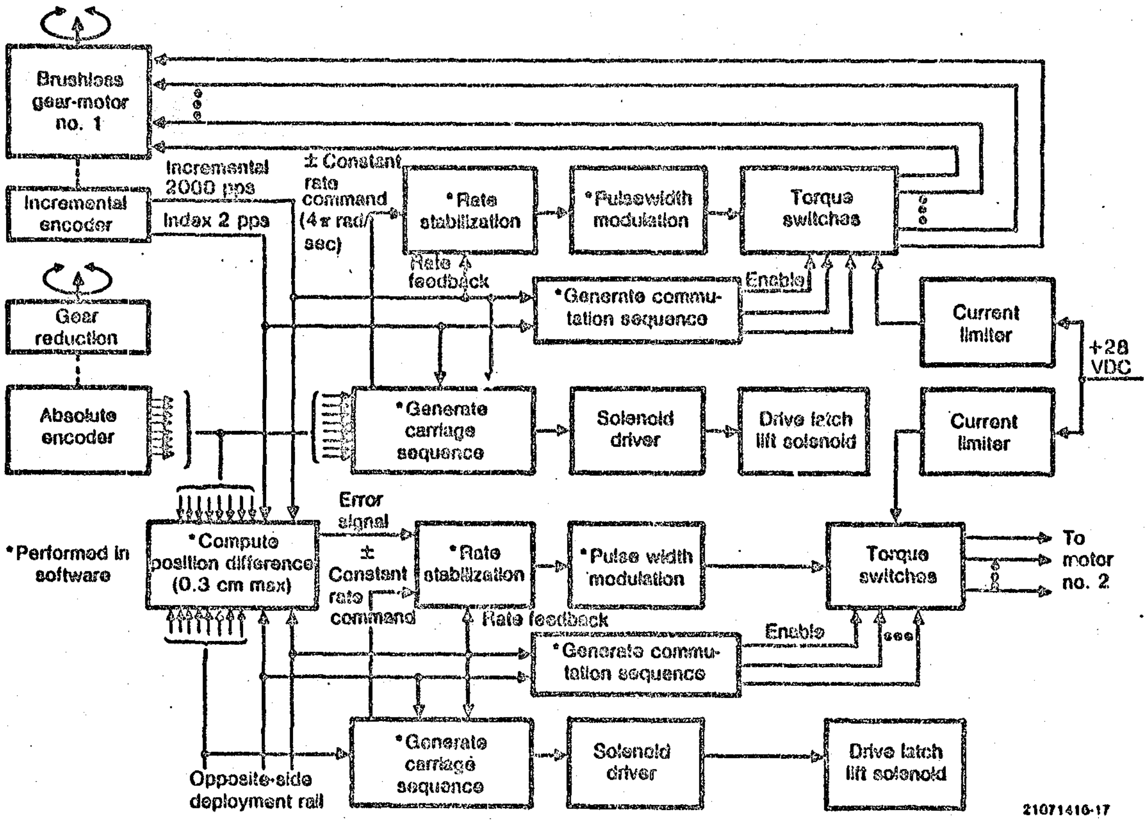

Figure 4-25. Carriage Control Electronics Diagram

The Payload Specialist can command or monitor any function performed by the Control Unit via the computer keyboard and display on the operator's panel, shown in Figure 4-26, with command and monitor software. He can command and monitor various experiment modes and excitation signals. He can monitor and assume control of the deployment/retraction operation. Redline indicators are provided for critical functions to immediately attract the Payload Specialist's attention. Continuous readouts are provided to indicate the extent of truss deflections and the progress of deployment or retraction. In case of emergency, the arm/safe switch can be operated for truss jettison.

The status of the jettison active fittings is displayed and also transmitted to the Orbiter C and W. After jettison, truss separation indicators show that a minimum separation distance has been achieved.

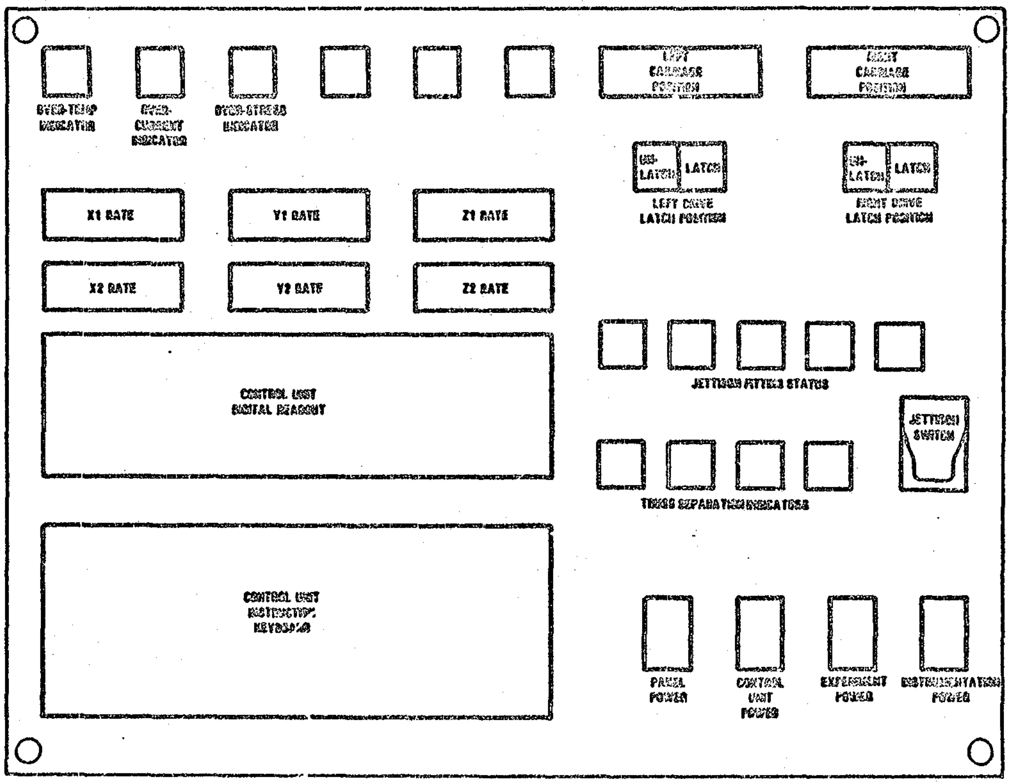

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Figure 4-26. IVA Operator Display and Controls Panel Concept

SECTION 5
ANALYSIS

Structural and dynamic analysis was performed to determine applied loads, support loads, and truss loads. These data were used to determine truss sizing and support geometry. Truss materials were evaluated and structural dynamic characteristics computed. Mass properties for the SCE were established and thermodynamic considerations evaluated. This section presents the results of these analyses.

5.1 STRUCTURAL ANALYSIS

The Convair prototype deployable truss (Figure 3-12) was configured to the size and strength requirements established by a previous LSS study of a large radar array. This configuration was used as the baseline structure for the SCE. The impacts of SCE operations on the baseline structural configuration were then determined.

5.1.1 APPLIED LOADS. The preliminary design of the SCE determined the maximum deployed length of truss was limited to approximately 50m by available stowage space. The size of the tip mass was selected to effectively double the roll moment of inertia of the Orbiter. Thus, a 400 kg tip mass would increase the roll moment of inertia by 10^6 kg-m^2.

The maximum on-orbit loads experienced by the truss and the truss support structure are caused by primary reaction control system (PRCS) thruster firings. PRCS thruster firings are considered worst-case contingency loads because they would normally not be used during space construction operations. However, during attitude control and maneuvering activities with the vernier RCS (VRCS), failure of a vernier jet to shut off may cause PRCS firings to occur.

The preliminary design loads were derived as quasi-static responses to PRCS firings. The steady-state responses caused by PRCS pitch and roll maneuvers were multiplied by a dynamic amplification factor of 2 for conservative estimates. The loads data were generated in the parametric form shown in Figure 5-1, assuming a rigid mounting interface.

Final design will require analysis of the selected SCE configuration in the Charles Stark Draper Laboratory (CSDL) DAP simulation to generate dynamic loads. The initial simulations performed by CSDL on an equivalent 100m truss with a 100 kg tip mass, however, did show a maximum dynamic load caused by a PRCS pitch maneuver that was approximately equal to the preliminary loads generated.

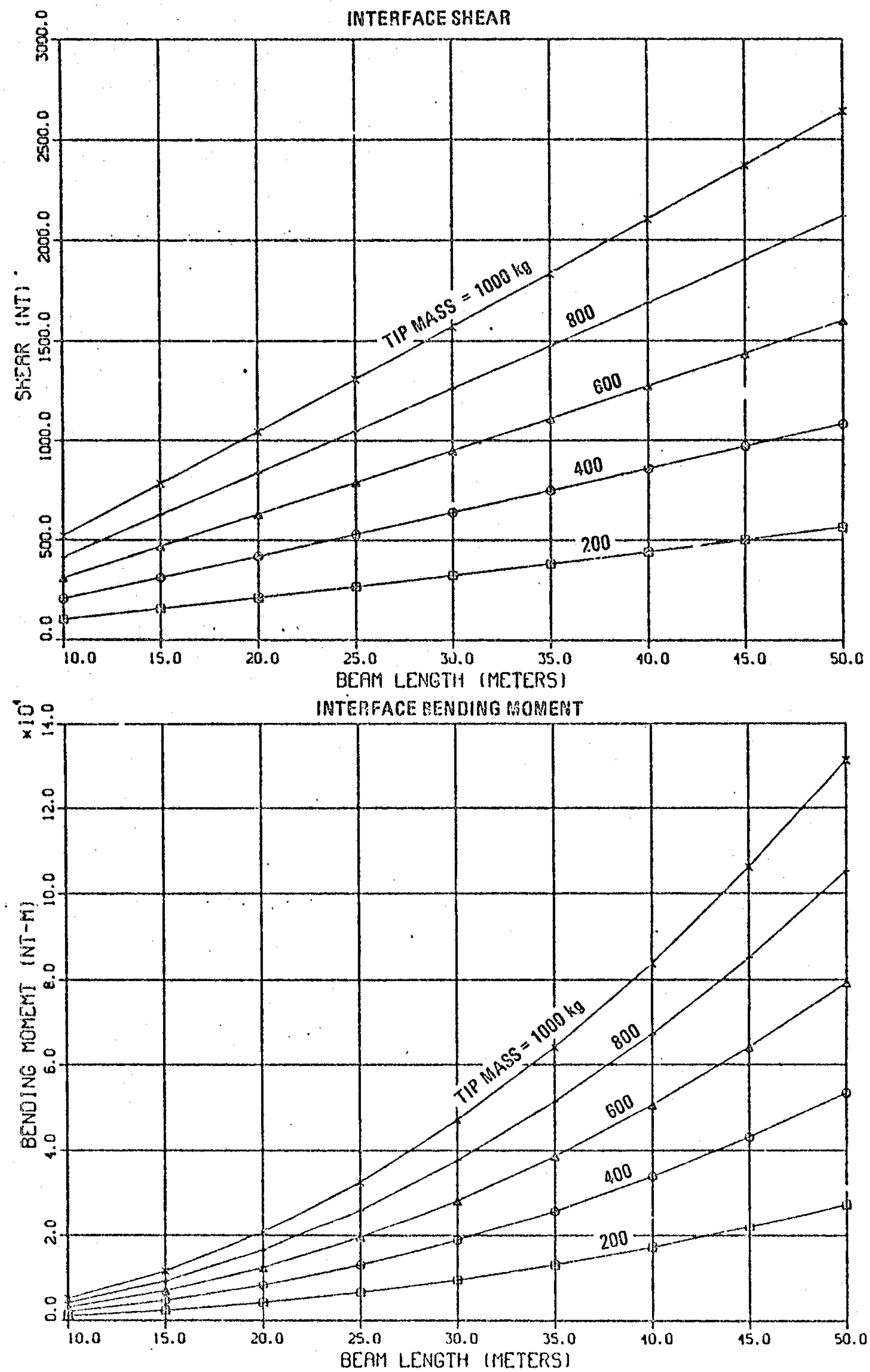


Figure 5-1. Parametric Truss Mounting Interface Loads Data

5.1.2 TRUSS SUPPORT. The stress analysis resulted in the truss support arrangement shown in Figure 5-2. The requirements include:

a. The rollers that support and guide the beam within the deployment rails must be captured to carry loads for both pitch and roll moments. Otherwise, deflections and loads in the rails are unacceptably large. Captured rollers must be configured to prevent binding during deployment.

b. Braces must be provided for the deployment rails. A single brace is needed for each rail to restrain pitching moments, while two braces per rail are required for roll moment restraint. Typical braces would be equivalent to the sizes shown using aluminum.

c. The deployment rails are 5.1 cm by 10.2 cm aluminum box beams with wall thickness of 0.5 cm. Margins of safety exceed 1.0 for worst-case loads.

The analysis considered the effects of various stages of deployment. The worst-case loads for roll moment occur with 22 bays and 27 bays deployed. In pitch, the worst-case rail loads occur with 22 bays deployed while maximum pitch brace loads occur at full deployment.

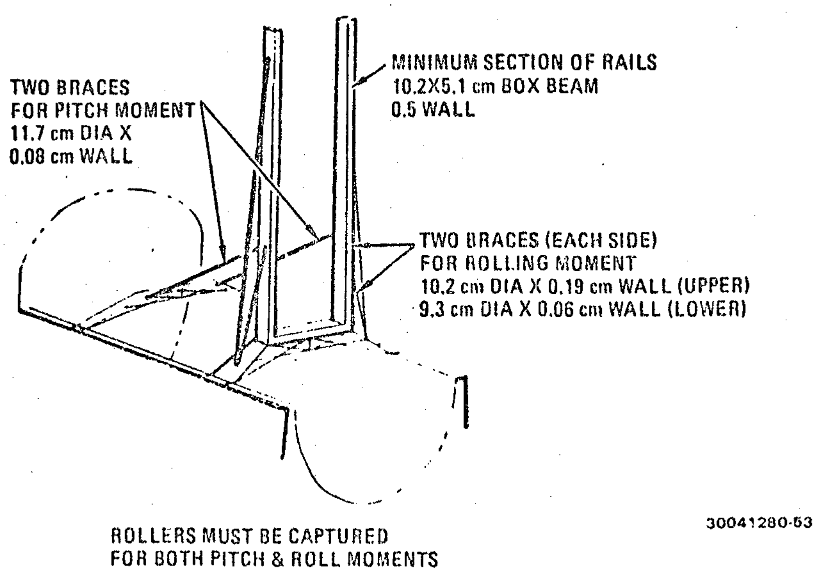


Figure 5-2. Truss Support Requirements

5.1.3 TRUSS LOADS AND GEOMETRY. The worst-case truss member loads are shown in Figure 5-3. The magnitude of these loads had a number of undesirable effects on the baseline truss geometry and configurations summarized in Table 5-1. The revised truss geometry and configuration is illustrated in Figure 5-4.

These physical changes increase weight, packaging size, and cost. They also increase the stiffness of the truss, which increases the modal frequencies. This detracts from the capability to perform DAP interactions testing as discussed in Section 6.3, where techniques for reducing truss stiffness are also discussed, and it is concluded that the best approach is to use flexible mounting.

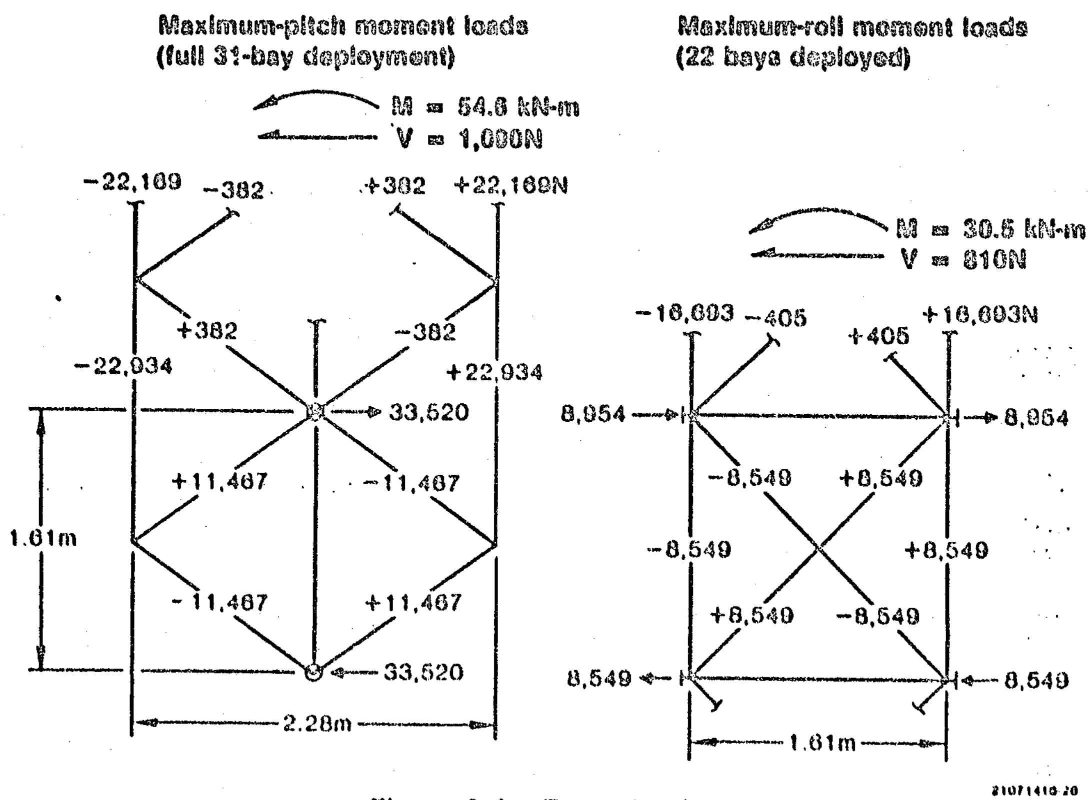


Figure 5-3. Truss Loads

Table 5-1. Deployable Truss Change Due to Contingency Loads

Item	Baseline	Change
Longitudinal Tube Diameter	4.50 cm	Same
Longitudinal Tube Wall Thickness	0.08 cm	0.13 cm
Diagonal Tube Diameter	2.54 cm	3.18 cm
Diagonal Tube Wall Thickness	0.08 cm	0.30 cm
1 Bay Packaged Length	17.42 cm	18.44 cm
Packaged Height	20.47 cm	23.67 cm
Diagonal Member Hinges	Carpenter Tape	Over-Center Locking

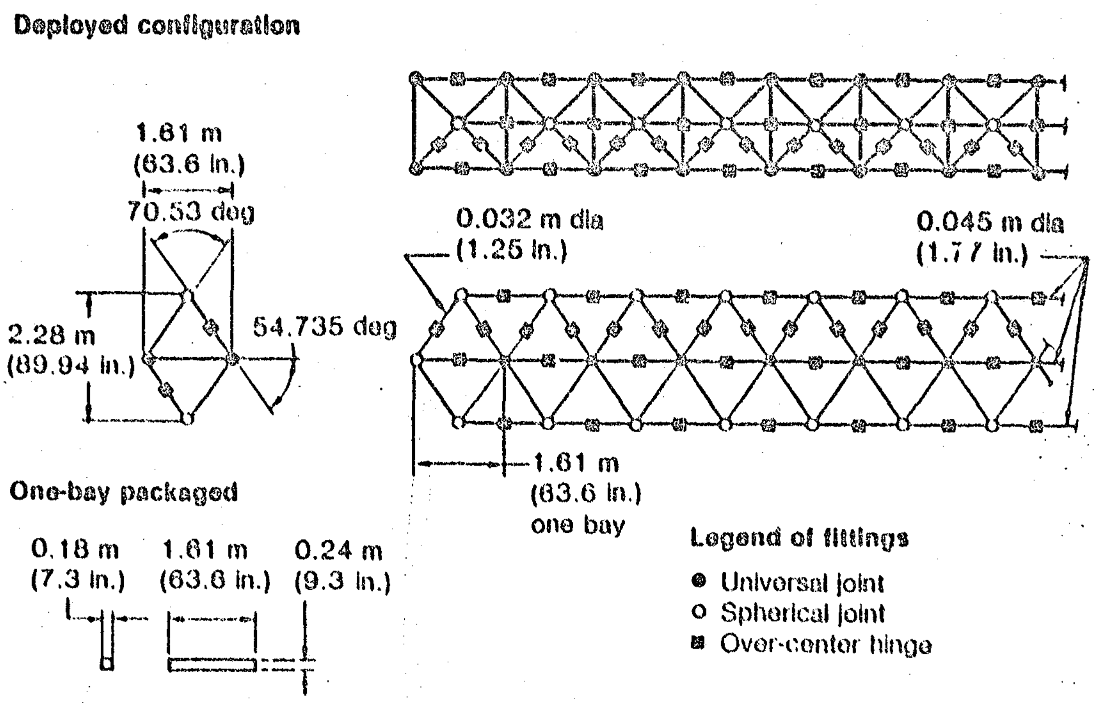


Figure 5-4. Revised Truss Geometry and Configuration

The modal frequencies of the truss are presented in Table 5-2 for two configurations. The first configuration assumes the truss and support rails are rigidly mounted to a rigid Orbiter. For this configuration, the lowest elastic mode is a roll bending mode at 2.0 Hz. The "soft mounted" configuration includes additional flexibility in roll between the truss and the Orbiter. The rotational spring value used was 1.0×10^5 N-m/rad. No additional flexibility was introduced in other directions. With the additional roll flexibility included, the first roll bending modal frequency dropped to less than 0.5 Hz.

The answer to the truss loads problem will come through further analysis where flexible mounting must be evaluated by DAP simulations. Once a mounting flexibility is selected and the size of the tip mass is validated, then the structure can be sized for optimum cost and performance.

Table 5-2. Comparison of Truss Modal Frequencies (Hz)

Mode	Rigid Mount	Soft Mount
1	0.200	0.046
2	0.237	0.237
3	2.96	2.00
4	3.67	3.67
5	8.71	7.34
6	10.15	9.24
7	11.94	11.94
8	18.54	15.62

<u>Truss Characteristics</u>

Length	50 meters
Tip Mass	400 kilograms
Stiffness (EI)	6.088×10^7 N-m^2 (pitch)
	2.936×10^7 N-m^2 (yaw)

5.1.4 <u>TRUSS MATERIALS EVALUATION</u>. The primary consideration for selection of materials and processes to fabricate a 50m deployable truss beam for a flight test experiment is cost. The Convair prototype deployable truss utilizes struts constructed of GY-70/930 graphite/epoxy. Mode fittings are aluminum and hinges are aluminum or titanium. Evaluation of the aluminum fitting joints showed the inadequacy of the aluminum-to-graphite coupling because of the gross difference in their coefficients of thermal expansion (CTE). The struts were fabricated by compression molding and were autoclave cured.

To produce a sufficient quantity of void-free quality tubing for the proposed 50m beam, two major production restrictions should be changed from the original fabrication scheme. These two restrictions are autoclave curing and compression molding. State-of-the-art tubing fabrication consists of layup on an aluminum male madrel, wrapping with shrink tape, and curing vertically in an oven. This method is currently in use to fabricate tubes for space applications that require void-free, dimensionally stable tubing.

The clamshell molded type of tube attachment fitting has become extremely popular. It offers potential cost savings compared to metallics such as titanium. The two metallic type fittings with acceptable CTEs to provide adhesive bond stability over a wide temperature range are titanium and Invar. High material cost and high machining cost restrict these metals as primary candidates. Carpenter tape hinges, however, require the use of titanium for their function.

The T-300 graphite fiber base fabrics are approximately 20% by weight of the UHM GY-70 fabrics and approximately 40% of the HM GY-50 fabrics. This makes T-300 a candidate only if low cost is considered. Its poor life and CTE stability characteristics and low modulus do not make it a viable choice. The high load and thermal stability considerations will dictate the use of the UHM GY-70 fabrics for the SCE truss.

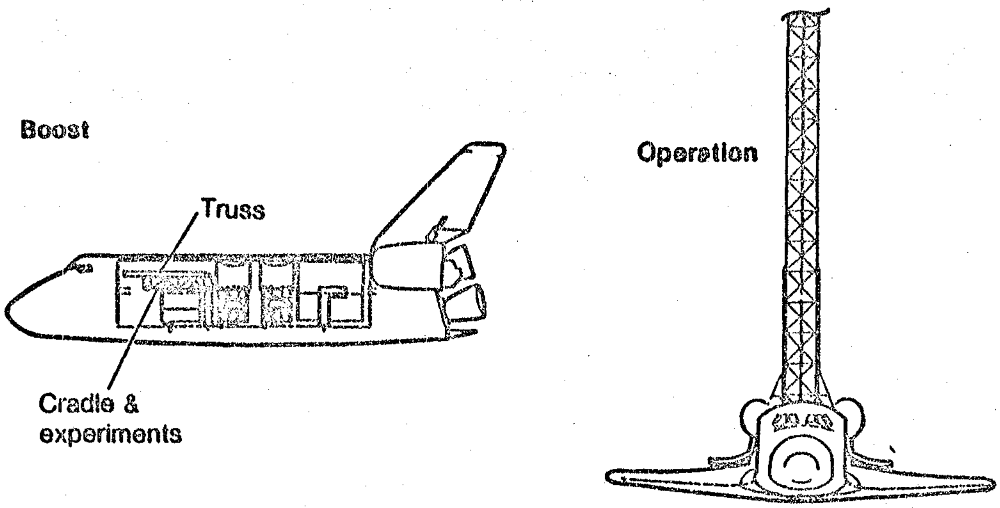
In summary, it is recommended that the GY-70/930 graphite-epoxy material be used extensively for both tubes and fittings. This will provide the best joint compatibility, minimize fitting manufacturing costs, and allow a near-zero CTE structure to be achieved.

5.2 MASS PROPERTIES

The mass properties computed for the SCE, including its support structure and experiments, are presented in Figure 5-5. The figures are given for the three planned phases of deployment for which dynamic and controls testing are planned. The moments of inertia are relative to the center of mass of the experiment. Mass properties of the Orbiter are not included.

A weight breakdown of the SCE is shown in Table 5-3. The tip mass is included as part of the truss equipment, and, as seen, it represents the major item of mass. Its presence accounts for the wide separation of the center of mass from the Orbiter. This is cause for concern where jettison of the experiment must be considered.

Use of the RMS to jettison the SCE while it is deployed could create unacceptable moments on the RMS. A system to first jettison the tip mass may be required.



Deployed phase	Center of mass (in.)			Moment of inertia (kgm^2)		
	X	Y	Z	I_{xx} (Roll)	I_{yy} (Pitch)	I_{zz} (Yaw)
1/3	775	0	668	8.11×10^4	8.21×10^4	2.53×10^3
2/3	775	0	958	3.03×10^5	3.04×10^5	2.53×10^3
Full	775	0	1,257	6.47×10^5	6.49×10^5	2.53×10^3

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Figure 5-5. SCE Mass Properties

Table 5-3. SCE Weight Breakdown

Item	Weight lb	Weight kg
Cradle	762	346
Truss	354	161
Deployment Structure	222	101
Deployment Mechanisms	156	71
Truss Equipment	955	433
Miscellaneous Electrical	35	16
Suitcase Experiments	200	91
TOTAL	2,684	1,219

5.3 THERMODYNAMIC CONSIDERATIONS

Thermodynamic considerations for the structures and mechanisms indicated in Figure 5-6 include selective thermal coating or shielding and grounding of electromechanical powered components such as motors and solenoids to prevent overheating. In areas where the astronauts may use handholds or otherwise contact the structure, thermal coatings will prevent excessively hot structure.

The issue of thermal deflections of the deployable truss is considered to be of minor importance provided the structural members are composite materials with very low CTE. Ground testing of CTE and heat transfer characteristics of truss struts and fittings is considered sufficient to accurately predict deflections. Specific measurement of thermal deflection in space is not planned, as these deflections will not be significant enough to warrant the added cost of measuring. Similarly, temperature measurements of the truss members would provide little useful data.

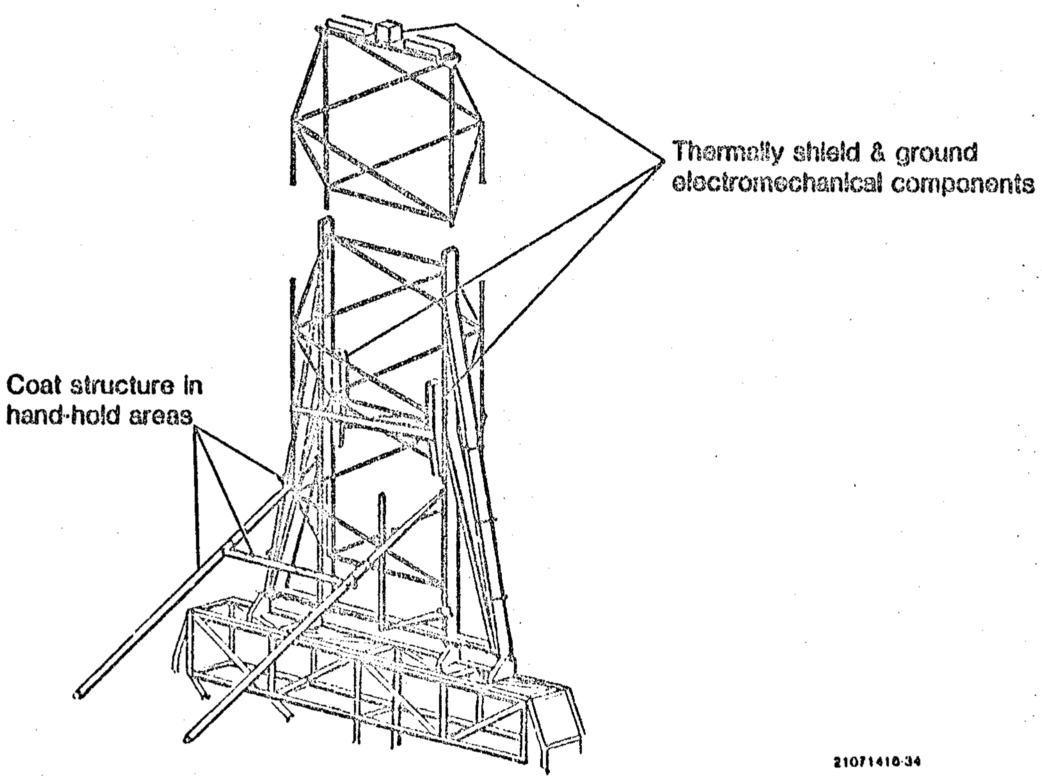


Figure 5-6. Thermodynamic Considerations

SECTION 6

FLIGHT CONTROL ANALYSIS

Studies and analyses were conducted by Convair identifying dynamic test requirements, damping requirements, and a selected damping approach. A preliminary NASTRAN model of the SCE was prepared by Convair, transmitted to the Charles Stark Draper Laboratory (CSDL), and integrated into their Digital Autopilot (DAP) simulations. Early simulation runs provided preliminary data on Orbiter/structure interactions and structural loads, and DAP performance provided data needed to identify SCE frequency and mode shape characteristics required to challenge the DAP. A second NASTRAN model of the preliminary design configuration with a range of base mount stiffnesses was then prepared by Convair and analyzed in CSDL's simulation. This section presents these study and analyses activities and results.

6.1 DYNAMIC TESTING

Structural and control dynamics tests were selected to evaluate key issues as identified in Table 6-1. The first test listed has been limited to roll maneuvers since that axis, with its smaller moment of inertia, is influenced much more by flexible structure than either pitch or yaw. Random noise modal surveys have been chosen since they are significantly more efficient time-wise than other techniques. However, one sinusoidal excitation and free decay test has been included to provide data on amplitude sensitive behavior.

Table 6-1. Dynamic Tests Related to Key Issues

	Issues			
TESTS	1	2	3	4
Small roll maneuvers at partial & full deployment — decreasing damping augmentation at each test length	●	●		
Random noise excitation modal surveys	●		●	●
Sinusoidal excitation & free decay of higher mode	●		●	●

Issues

1. Effect of test structure flexibility & vibration on Orbiter & DAP
2. Effect of Orbiter-induced dynamics on test structure
3. Minimum modal damping ratios
4. Dynamic modeling accuracy, especially for higher modes

One dynamic issue that has not been included is the effect of outriggers or offset feed provisions on the end of the beam. Noting that any structure is dynamically modeled by mass properties and mode, the preliminary conclusion is that the added structure does not present a different dynamic problem: the incremental moment of inertia contributions are small and the first bending mode would not be significantly different in the critical roll axis.

Excitation of the structure for the modal surveys and sinusoidal excitation tests is provided by the torque wheels located in the damper sets at the tip of the test structure as discussed in the subsequent subsection.

The rate gyros in the damper sets will be used to measure the lower modes, while distributed accelerometers (approximately 15) will be used to measure higher frequency mode shapes and damping.

Bending loads in the truss will be measured by strain gauges installed on truss members in selected bays at the base of the deployed truss. Orbiter/structure interaction loads will be measured by load cells in the mounts and supports of the truss deployment rail.

6.2 DAMPING AUGMENTATION

To evaluate how much damping augmentation will be required, the behavior of a 100m truss with a 100 kg tip weight was analyzed. The truss was cantilevered from a rigid body Orbiter. Since this structure is longer than any of the experiment candidates, it represented an extreme case such that conclusions reached would be conservative. This preliminary model had these characteristics:

a. Pitch natural frequency = 0.96 rad per sec

b. Assume first bending mode damping ratio = 0.001

The use of the 0.001 damping ratio may not be conservative. As part of Convair's IRAD work, modal damping ratios of 0.0025 and lower in air have been measured.

Two requirements were identified and evaluated for various values of first bending mode damping ratio. The first requirement is to minimize the test structure response to Orbiter-induced excitation. The structural responses for 0.1%, 1%, and 2% damping ratio were computed assuming the Orbiter pitching in a ±0.01 degree per second square wave with a frequency of 0.96 radian per second. The values are shown in Table 6-2.

The second requirement is to limit the time required to stabilize the structure prior to performing a subsequent test. Times for the selected damping ratios are also shown in Table 6-2.

Table 6-2. Variable Damping Ratio Effects on Tip Motion and Stabilization Time

Damping Ratio	Steady-State Tip Motion, meters	Minutes to Stabilize
0.001	±11.4	87.0
0.01	± 1.2	8.7
0.02	± 0.6	4.3

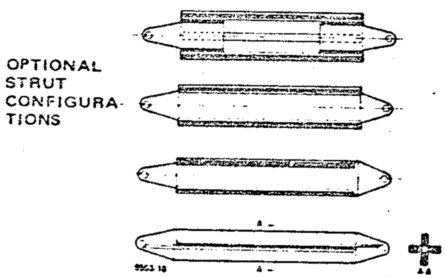
Based on the predicted responses and damping times for the preliminary model, it was concluded that 1% damping with augmentation may be sufficient and 2% damping is adequate, depending of course on the final configuration selected. For the first requirement, it should be noted that the amplitude buildup for lightly damped modes is a rather slow process even when the excitation is at the exact critical frequency.

Three damping augmentation approaches were considered for the SCE. The alternative approaches are shown and compared in Figure 6-1.

Viscoelastic materials were rejected since creep and outgassing require further development efforts. The proof-mass/accelerometer damper is an excellent technique for high frequencies but becomes severely stroke limited at low frequencies. Applying 6.28 rad/sec to the output relation shown in Figure 6-1 gives about 5 Newtons (N) available at 1.0 Hz rather than the rated value of 196 N. Low frequencies, as will be encountered in the first mode of the test structure, are best damped with the torque wheel/rate gyro damper, as used with dramatic success on a Convair IRAD program. Although the IRAD wheel starts limiting at 1.3 Hz, this frequency can be readily reduced by additional weight and size.

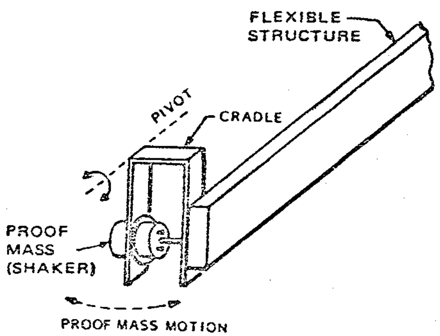
The installation concept for the selected damping augmentation approach is shown in Figure 6-2. By using two torque wheel damper sets per axis with each set providing 1% damping to the first bending mode, it is possible to select 2% damping (both sets operating), 1% damping (one set operating), or zero added damping with both sets off. Sizing the maximum torque of the wheels is not especially critical since they still provide damping in saturation but not as much as when they are operating in the linear range. The installation shown includes provision for variable tip masses by pumping fluid into closed cylinders. Thus, between partial deployment and partial tip mass, the extreme condition can be approached in fine increments. Preliminary sizing indicates a maximum torque of 4.5 Nm as set by a 50m truss and a 0.05 deg/sec step change in Orbiter body rate.

Passive damping with viscoelastic materials



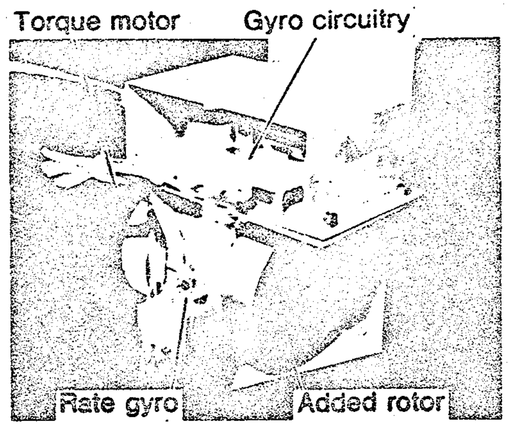
- Creep & outgassing are unresolved issues
- Application would negate basic structural damping evaluation

Proof-mass/accelerometer damper



- Consider Ling 403 shaker as proof-mass actuator
- Output = 196N (44 lbf), stroke = ±0.88 cm, 24 cm dia × 39 cm lg, 14 kg
- Stroke limits low frequency output to $\omega^2/8$ N

Torque wheel/rate gyro damper ✓



- Torque wheel is best low frequency actuator known
- IRAD torque wheel has full output from 1.3 Hz to 100 Hz
- Suitable torque wheels can be obtained by using larger motors in existing flightworthy units

Figure 6-1. Candidate Damping Augmentation Approaches

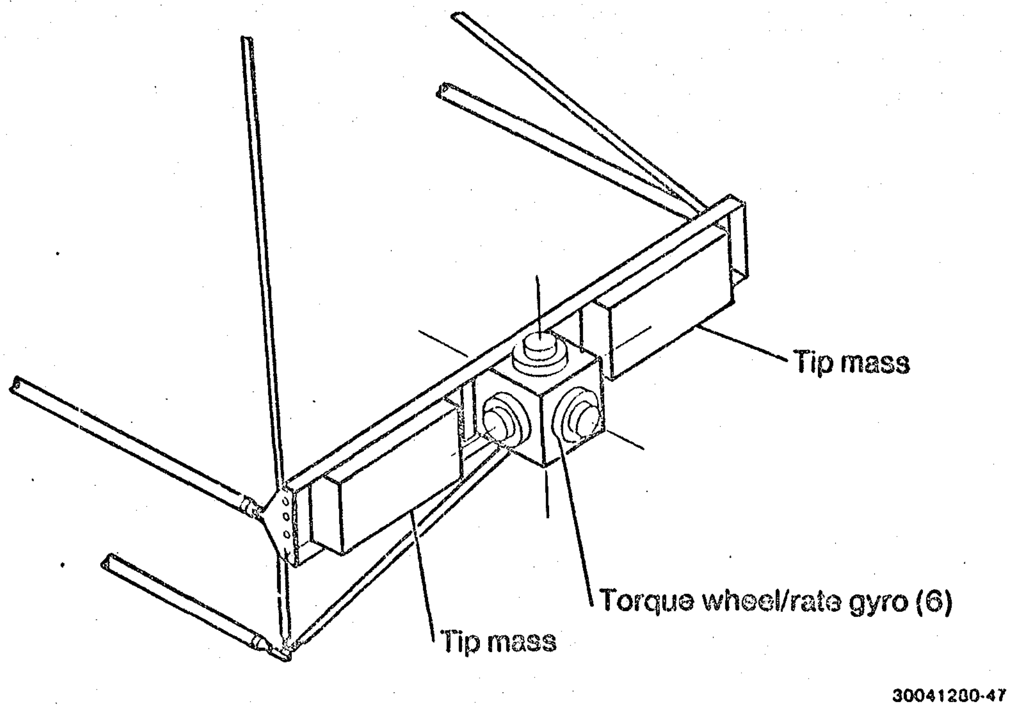


Figure 6-2. Selected Damping Augmentation Approach

Calculations early in the study used an active damping component set with a high bandwidth torque wheel and a rate gyro that has a natural frequency of 20 Hz. The rate gyro dynamics limited performance since higher frequency modes could be driven unstable at high values of gain. Later developments removed this limitation by blending the rate gyro output with that of a high frequency sensor such that the component set is flat to about 800 Hz. As seen in Table 6-3, a first order is then added well below the 800 Hz (in this case at 40 Hz or 250 rad/sec); consequently, the system can roll off without introducing sufficient phase shift to cause stability problems. The data show that for the 1% or 2% damping ratios of interest here, the gain margin is at least a factor of 5 (14 dB).

6.3 DAP CONSIDERATIONS

Possible interactions between the DAP and the SCE structure can come from a number of sources. The most complex interaction arises from large flexible structure with low modal frequencies and large moment of inertia contributions. As the structure grows larger and the frequencies get lower, there must be a limit to the ability of the current DAP to maintain control. Attempts to identify and understand this limit have been given considerable emphasis in this program.

Table 6-3. Active Damper Performance for 50m Truss with 400 kg Tip Mass using Wideband Torque Wheel-Sensor Combination with First Order Filter at 250 rad/sec

Mode	Freq Rad/Sec	Axis	Damping Ratio Increase K=1	K=2	K=5.7	K=11.4
1	1.35	P	0.014	0.028	0.082	0.16
2	1.91	R	0.010	0.020	0.057	0.12
3	40.6	P	0.077	0.15	0.28	0.15
4	57.4	R	0.050	0.10	0.24	0.20
5	67.9	R	0.033	0.062	0.11	0.89
6	130.9	P	0.057	0.11	0.13	0.072
7	185.2	R	0.016	0.032	0.086	0.13
8	272.9	P	0.036	0.076	0.18	0.19

K = Feedback gain

DAP performance is measured by two criteria. The first is the rate of RCS propellant consumption. The second is achievable pointing accuracy. Assuming that the SCE structure will be designed to preclude damage from worst-case RCS firing conditions, the other possible performance limiters include:

a. Deployed structure flexibility

b. Deployment transients

c. RMS operations

d. Products of inertia

e. Center-of-mass shifts

6.3.1 CSDL COMPUTER SIMULATIONS. A preliminary NASTRAN data tape was prepared at Convair and sent to CSDL. In addition to checking out the data transmission interface, it was expected that computer simulation of the DAP and the structure described by the data would provide some information on the DAP performance limit. The preliminary tape includes a model of a 100m truss without tip mass and a 100m truss with a 100 kg tip mass.

The results of the CSDL simulations and analysis are documented in Reference 27. Table 6-4 summarizes the simulation runs made at CSDL. The simulations included one manual control case and two cases with primary RCS jets failed on. Although the 100m beam with the 100 kg tip mass gave larger moment of inertia changes than any payload previously run at Draper, the conclusion was that the DAP could handle it without any significant performance degradation.

Table 6-4. Simulation Run Summary for Preliminary 100m Truss with 100 kg Tip Mass

	VRCS			PRCS		
Maneuv Axis	Maneuv Angle	Rate Limit	Deadband	Maneuv Angle	Rate Limit	Deadband
Roll	10 deg	0.02 deg/sec	1 deg	40 deg	0.3 deg/sec	5 deg
Pitch	10 deg	0.02 deg/sec	1 deg	40 deg	0.3 deg/sec	5 deg
Yaw	10 deg	0.02 deg/sec	1 deg			
R,P&Y	10 deg	0.02 deg/sec	1 deg			
Roll	10 deg	0.02, T > 60 0.01, T < 60	1, T < 60 0.1, T > 60			

Maneuver Rate: 0.25 deg/sec VRCS; two deg/sec PRCS

6.3.2 FLEXIBLE PAYLOAD COMPARISONS. In an attempt to better understand the flexible payload/DAP interactions, data were assembled on other payloads simulated at CSDL. These data are summarized in Table 6-5. Since the deployed/stowed inertia ratio is indicative of the effective mass associated with the flexibility, it is significant to note that the only payload that showed signs of having a DAP problem had one of the lowest ratios. That was the RMS-PEP, which displayed a significant (21%) increase in RCS propellant consumption. However, RMS-PEP did have the lowest bending frequency.

Considering the facts available, it appeared there was some frequency-sensitive element in the system that attenuated the structure-induced oscillations of the Orbiter before they reached the jet logic.

Table 6-5. Flexible Payload Comparison

Payload	Lowest Frequency (hertz)	$\frac{\text{Deployed } I_{xx}}{\text{Stowed } I_{xx}}$
RMS-PEP	0.052	1.19
Space Telescope	0.566	1.20
IUS/TDRS	0.127	1.18
IUS/Galileo	0.16	1.36
IUS/DoD1	0.097	1.25
Beam, 100m, 100 kg	0.14	2.00

6.3.3 DAP STATE ESTIMATOR. The DAP State Estimator was identified to be the frequency-sensitive element in the DAP that was attenuating the beam-induced oscillations. The filter characteristic shown in Figure 6-3 is for the default filter gains - those gains the computer uses unless other values are specified. The 0.05 Hz oscillations of the RMS-PEP were cut in half by the filter, and this relatively low mass payload still caused a moderate increase in propellant consumption. It would appear that heavier payloads with a bending frequency of 0.05 Hz or less may have severe problems with the filter as shown. Of course, the filter could be changed so that it started cutting off at a still lower frequency, but while this would eliminate flexibility problems there could be other problems caused by the rate information being too old by the time it reaches the phase plane logic. This is an area for further study. However, it is clear that if the SCE structure is to evaluate the proven limits of the DAP, the first mode bending frequencies must be lowered.

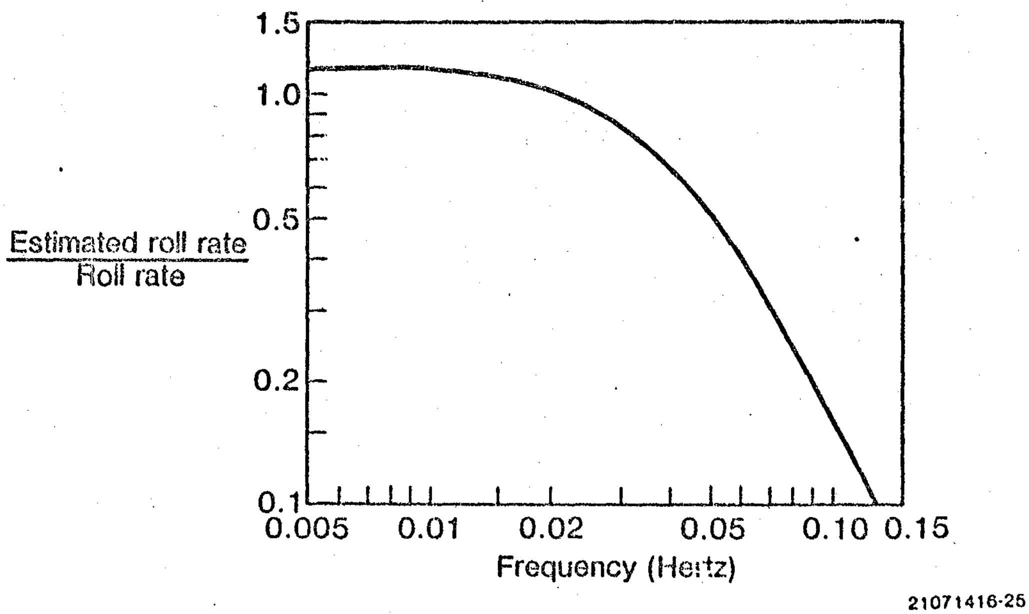


Figure 6-3. State Estimator Filter Characteristics

6.3.4 REDUCING BENDING FREQUENCY. The following techniques were considered to reduce the bending frequency of the SCE structure:

a. Employing a longer truss is not feasible due to stowage space limitations.

b. Reducing the cross-section geometry and moment of inertia of the truss requires iterative analyses to optimize size and loads capabilities for frequency requirements.

c. Providing a tapered cross-section geometry would significantly increase the cost and complexity of the deployable structure.

d. Increasing the tip mass means increasing the strength of the truss members, which further increases truss stiffness.

e. Changing material to a lower modulus material would also reduce load carrying capability of the truss.

f. Use of a flexible mounting scheme for the deployable truss would tend to relieve bending loads in the truss. Base mount flexibility can also readily be adjusted during analysis and would have minimum impact on the final design should late changes be required. An initial look at the effects of flexible mounting on bending frequency showed that for the 50m truss with a 400 kg tip mass, the first roll bending mode was reduced from 0.200 Hz to 0.04 Hz with a mounting stiffness of 10^5 Nm/rad.

The use of a flexible mount was selected to reduce SCE bending frequency because it is the only approach identified that has no undesirable features or limitations. Since the flexibility will tend to relieve loads, it is also a good candidate for operational large structure use because it will minimize the strength required while the structure is in the attached mode.

SECTION 7
PRELIMINARY TEST PLAN

7.1 INTRODUCTION

The Space Construction Experiment (SCE) is a basic early Shuttle flight experiment that will develop and test the capabilities of the Space Shuttle system to support construction of large space systems. The basic SCE will consist primarily of a large deployable structure equipped with controls, instrumentation, and representative subsystems elements to allow testing of Orbiter control during and after construction, construction operations using basic Orbiter capabilities, and predicted dynamic behavior and control of a large deployable structure attached to the Orbiter.

The SCE will be integrated into the Shuttle as a secondary payload of opportunity with testing to be performed on a noninterference basis with primary payloads.

7.2 PURPOSE OF THE TEST PLAN

The purpose of this preliminary test plan is to define the requirements for development tests, ground tests, and flight tests of the SCE and to describe the test operations, test sequences, and instrumentation concepts for the SCE program.

7.3 GROUND RULES AND ASSUMPTIONS

The SCE test program shall be conducted in accordance with the following ground rules and assumptions:

a. Flight qualification testing will be primarily performed at the system level to minimize the cost of verifying overall flight worthiness of the experiment.

b. Major flight qualification tests are planned using JSC facilities.

c. The flight test operations will be conducted at KSC and aboard the STS Orbiter.

d. Only one test article shall be produced for ground and flight testing.

e. The development test program may be initiated prior to Phase C/D to perform preliminary investigations and ground test simulations using prototype and mockup hardware.

7.4 TEST PROGRAM SUMMARY

The preliminary test program plan is summarized in Figure 7-1. The development testing phase will allow system requirements to be defined for the program Phase C/D design and development effort. The flight qualification tests will verify the flight worthiness, environmental compatibilty, and functional capability of the SCE. Plans for flight test follow the flow as shown.

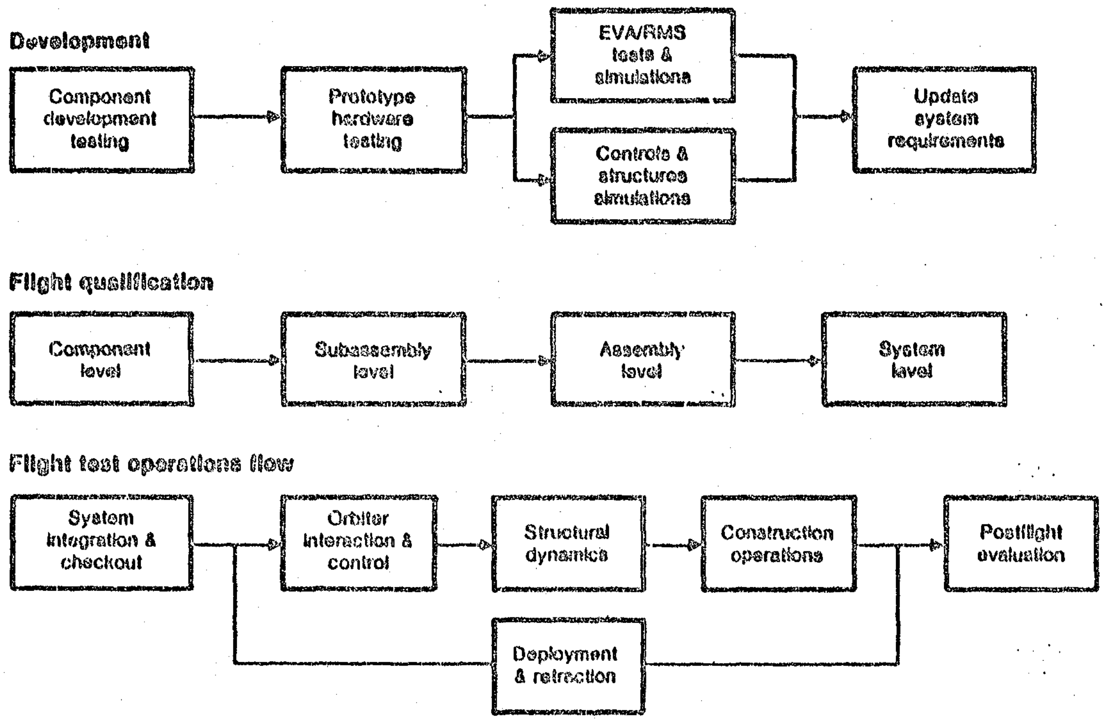


Figure 7-1. Top Level Test Program Summary

7.5 TEST PLAN

7.5.1 DEVELOPMENT TESTS. The preliminary development test program is summarized in Table 7-1. Early investigation and development of components will be required to support the anticipated short-term program span. Adapting existing flight-qualified torque wheels and rate gyros to this application will be a long-lead-time consideration. Manufacturing of the deployable truss will be a major cost driver and will require some technology development to achieve a cost-effective design.

Early tests utilizing a prototype truss segment will ensure compatibility of the final truss design with the operational environment (handling, construction, transportation). It will also allow structural test techniques to be developed for correlation of full-scale assembly dynamics with analysis and subassembly tests.

Table 7-1. Preliminary Development Test Program Summary

Program elements	Key development issues	Requirements/Objectives
• Components		
— Torque wheel/rate gyro	— Adapt qualified components	— Demonstrate space-compatible damper
— Truss tubes & fittings	— Materials & producebility	— Low-CTE, low-cost structural performance
— Attachment fittings	— Adapt available technology	— Integrate into structure
• Prototype hardware		
— Truss segment & elements	— Operational compatibility — Predictive test techniques	— Support EVA/RMS ground tests — Space structure performance prediction
• EVA/RMS tests & simulations	• Construction operations safety & efficiency in simulated space/orbiter working environment • Flight experiment human factors & design requirements • Timelines & procedures	• Develop manual tools & techniques • Develop suitcase experiment candidates • Develop RMS-aided deployment & assembly techniques & tools • Develop restraint, illumination & RMS handling devices • Develop stowage arrangements & techniques
• Controls & structures simulations	• Enhanced computer simulation	• DAP simulation analysis of combined flexible orbiter/flexible structure

EVA/RMS tests and simulations will be required to develop the complete experiment package (suitcase experiments, tools, accessories, and procedures). These tests will be performed at JSC's Manipulator Development Facility (MDF) and Weightless Environment Test Facility (WETF).

The capability to perform DAP computer simulations of the combined flexible Orbiter and flexible test structure will allow a greater level of refinement in the design of the experiment and greater confidence in the predicted Orbiter/ experiment behavior.

7.5.2 FLIGHT QUALIFICATION TESTS. The flight qualification test program is structured to ensure the safety and functional reliability of the SCE at minimum cost. Environmental testing is minimized by performing major tests at the integrated system level to preclude numerous individual component and subassembly tests. The test levels, test specimens, and tests required are shown in Table 7-2 and are described as follows:

a. Component level tests will be conducted on the structured truss element. This includes struts, hinges, and fittings. Thermal vacuum tests of struts and fittings will be performed to verify heat times for characteristics of components to support thermal modeling and joints. Thermal cycling tests will verify bonded joint integrity and thermal stability characteristics. Humidity tests will verify moisture resistance characteristics of composite material components for outgassing considerations.

Table 7-2. Preliminary Flight Qualification Test Program Summary

Environment / Item	Ambient operating	Vacuum or thermal vacuum	Vibration	Acoustic	EMC	Shock	Climatic
Component level							
• Truss elements	Structural	Heat transfer					Thermal cycling & humidity
Subassembly level							
• Damper package	Phasing	X			X		
• Truss segment	Structural						
• Deployment carriage	X	X			X		
• Controls	X		X		X		
Assembly level							
• Deployable truss assembly	X					X	
Integrated system							
• Truss assembly, cradle & experiments	X	X	X	X	X		

b. Subassembly level tests will be conducted on:

1. The damper torque wheel/rate gyro package will be functionally tested in ambient conditions to set up the phasing. The package will be tested for EMC and subjected to a functional thermal vacuum test.

2. Structural tests of a production segment of deployable truss will be conducted to measure stiffness and damping characteristics to support structural dynamic modeling.

3. The deployment carriage will be run through a long series of operating cycles in thermal vacuum to confirm its durability and reliability. It will also be tested for EMC.

4. The aft deck controls panel and controller package will be functionally tested by supporting the damper and carriage tests. They will also be vibration tested individually and EMC tested.

c. An assembly level test of the completely assembled deployable truss and deployment mechanism will be conducted. The truss will be deployed and retracted horizontally with the aid of support rollers/cables. The packaged assembly complete with tip mass and damper package will be shock tested.

d. The assembled SCE payload will be delivered to JSC and tested in the sequence shown in Figure 7-2. Ambient tests and operations to be conducted in the MDF include:

1. Crew training exercises
2. RMS aided deployment exercises
3. Partial automatic deployment/retraction exercises
4. Experiments installation, test, and stowage exercises
5. Problem solving and flight test procedures updating

EMC test will be conducted at ambient conditions, followed by acoustic, random vibration, and thermal vacuum test of the SCE truss, cradle, and experiments package.

7.5.3 <u>FLIGHT TEST PREPARATIONS</u>. Following flight qualification testing at JSC, the SCE will be shipped to KSC for flight test preparation activities. These will include:

a. Off-line preparation including refurbishment, final instrumentation installation, and checkout.

b. Level IV integration of experiment, cradle, and controls to verify mechanical/electrical interfaces, payload to Orbiter interfaces using the Cargo Integration Test Equipment (CITE), and EMI/EMC compatibility testing.

c. Level III/II integration to prepare the SCE for final installation into the Shuttle. This includes installation of this real protection hardware, installation of experiments on the cradle, and final securing of the test truss to the cradle.

d. Level I installation of the SCE in the Orbiter cargo bay and installation of interface hardware and aft flight deck controls. This includes final checkout of controls and instrumentation and validation of electrical interfaces.

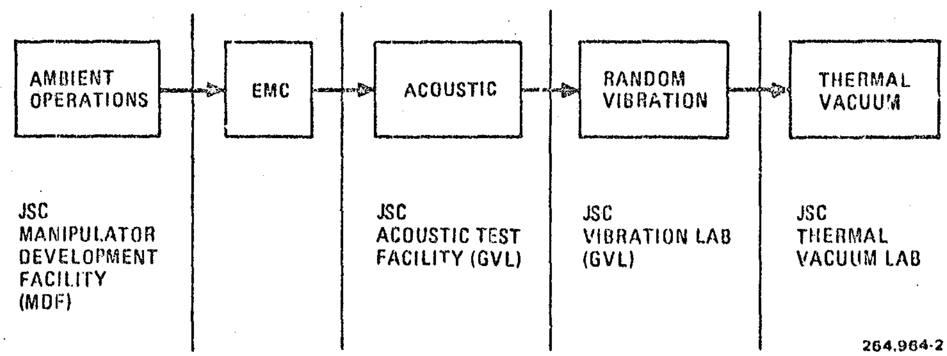


Figure 7-2. Qualification Test Sequence

7.5.4 FLIGHT TEST/MISSION OPERATIONS. The flight test sequence will require two days of the total mission. The first few days in orbit will be used to deploy the satellite payloads. Following these operations the SCE activities will be initiated.

The flight test operations sequence and timelines for the first day of the experiment are shown in Figure 7-3. The first day's activities will include a series of controls and dynamics tests primarily aimed at verifying or defining the limits and characteristics of the Orbiter DAP. The major test sequences are described in Figure 7-4.

The day one experiments will be conducted by the payload specialist and the pilot and/or mission specialist working at the aft flight deck control and display panels. The payload specialist will control and monitor the experiment while the pilot performs RMS operations and controls the roll maneuvers.

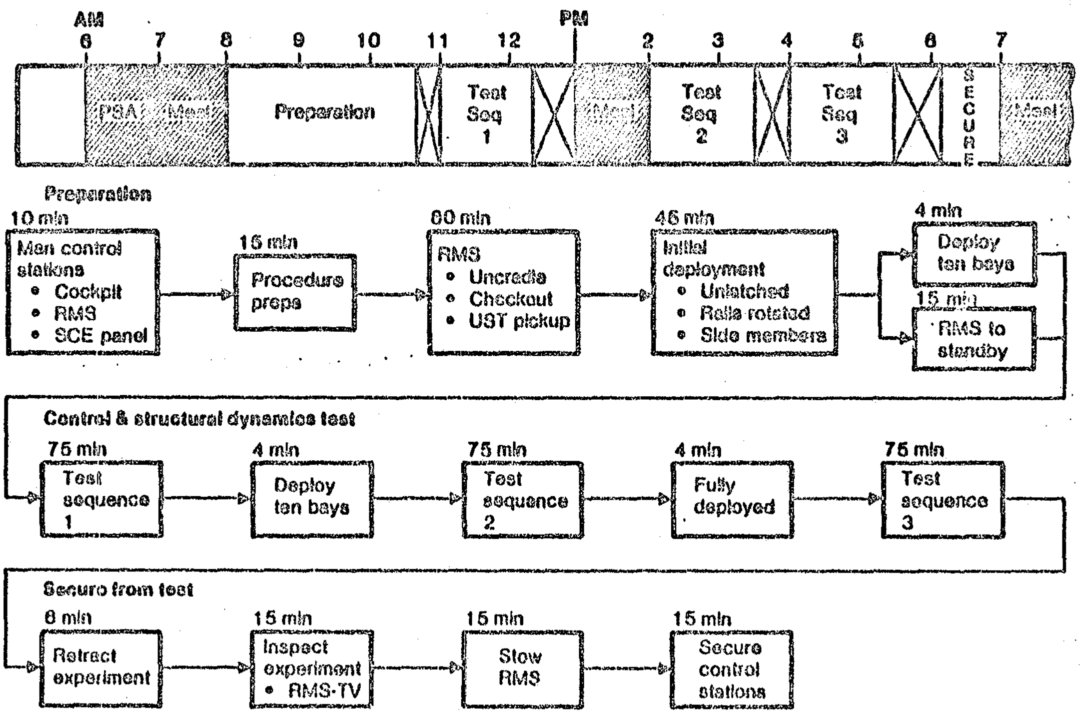


Figure 7-3. Flight Test Operations Sequence and Timelines for Day 1

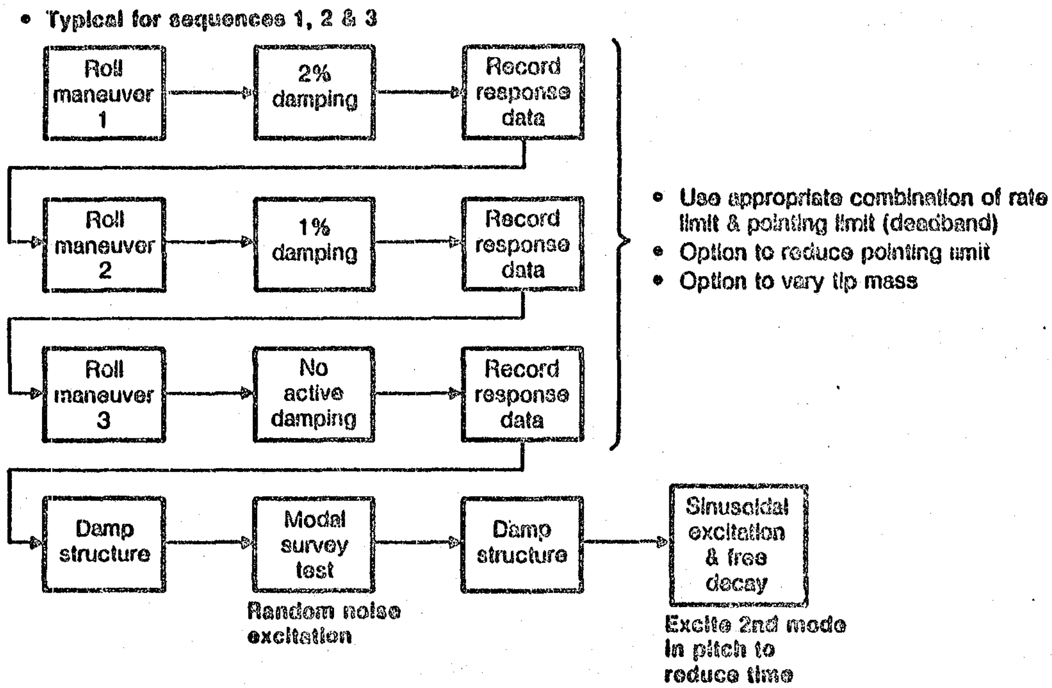


Figure 7-4. Structural Dynamics and Control Test Sequence

The objectives of the structural dynamics and control test sequences are:

a. Determine the effect of the test structure flexibility and vibration on the Orbiter and DAP.

b. Determine the effect of Orbiter-induced dynamics on the test structure.

c. Measure minimum modal damping ratios.

d. Test dynamic modeling accuracies, especially at higher modes.

The test sequence is designed to approach the effects on the DAP in a cautious manner by starting at partial deployment and incrementally reducing damping. If necessary, provisions can be incorporated to vary the tip mass and pointing limit.

The construction operations test sequence will be conducted on the second day of the experiment. This test sequence, illustrated in Figure 7-5, includes several assembly and installation tasks that require manual and EVA-assisted operations. The EVA tasks will be performed by the mission specialist and the commander. The payload specialist will continue to control and monitor the SCE from the aft flight deck control and display panel, while the pilot performs the RMS operations. The EVA will remain in effect until all equipment is fully stowed for reentry and landing.

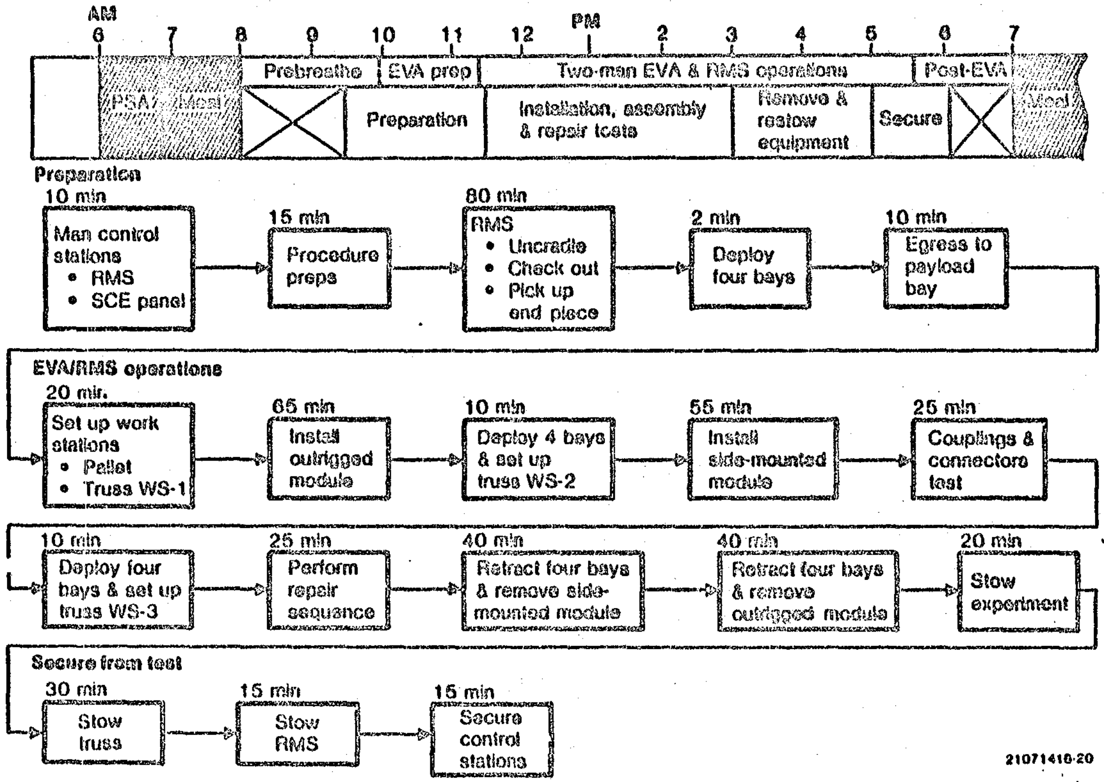


Figure 7-5. Construction Operations Test Sequence and Timelines for Day 2

As seen from the timeline, the actual amount of time available to perform construction operations is limited by the preparation, removal, and restowage and securing time. The rationale for the operations selected is discussed in Section 3.2.

7.5.5 POST-MISSION OPERATIONS. After descent, landing, safing, etc., the SCE will be inspected visually for evidence of hard use or damage. It will be removed from the Orbiter and transported to a designated site for further evaluation.

Data tapes of recorded flight experiment measurements will be transmitted to the contractor's facilities for reduction and analysis.

7.6 OPTIONAL SPACE CONSTRUCTION EXPERIMENTS

7.6.1 OPTION CONCEPT 1. As an alternative to performing the series of EVA/RMS operations described in Section 7.5.4, an experiment that would maneuver and connect a truss member to a truss could be performed, as shown in Figure 7-6. This type experiment could be accomplished within cost guidelines for the SCE since it requires little additional hardware to perform. The advantages of performing this experiment are:

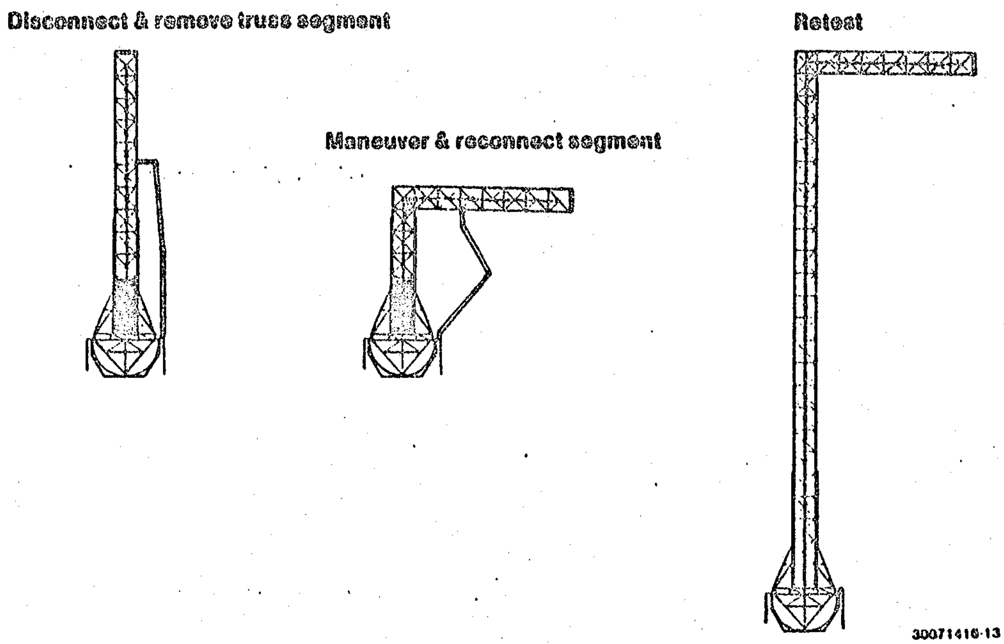


Figure 7-6. Optional Construction Experiments Concept 1

a. It would be similar to the mass properties represented by an antenna reflector mounted to a long feed mast, either by edge mounting or from a support arm. This would allow dynamic characteristics of that type of arrangement to be tested.

b. It would test the capability of the RMS to maneuver and position a piece of flexible structure.

c. It would allow an in-space demonstration of a truss-to-truss joining operation to be performed.

d. The disconnection and reconnection of service lines would be demonstrated due to the presence of electrical and possibly fluid lines along the length of the truss.

7.6.2 OPTION CONCEPT 2. A second alternative construction operations experiment that would provide a greater challenge to the manned construction activities sequence would be to employ two to four deployable/retractable coiled truss assemblies to produce a planar structure as shown in Figure 7-7. This more complex structural arrangement would demonstrate a higher degree of assembly operations complexity. The planar structure would also cause the vibration modal spectrum to become more densely populated. This would enhance the DAP effects testing by causing the Orbiter to see more modes in the lower frequency range where it is most sensitive.

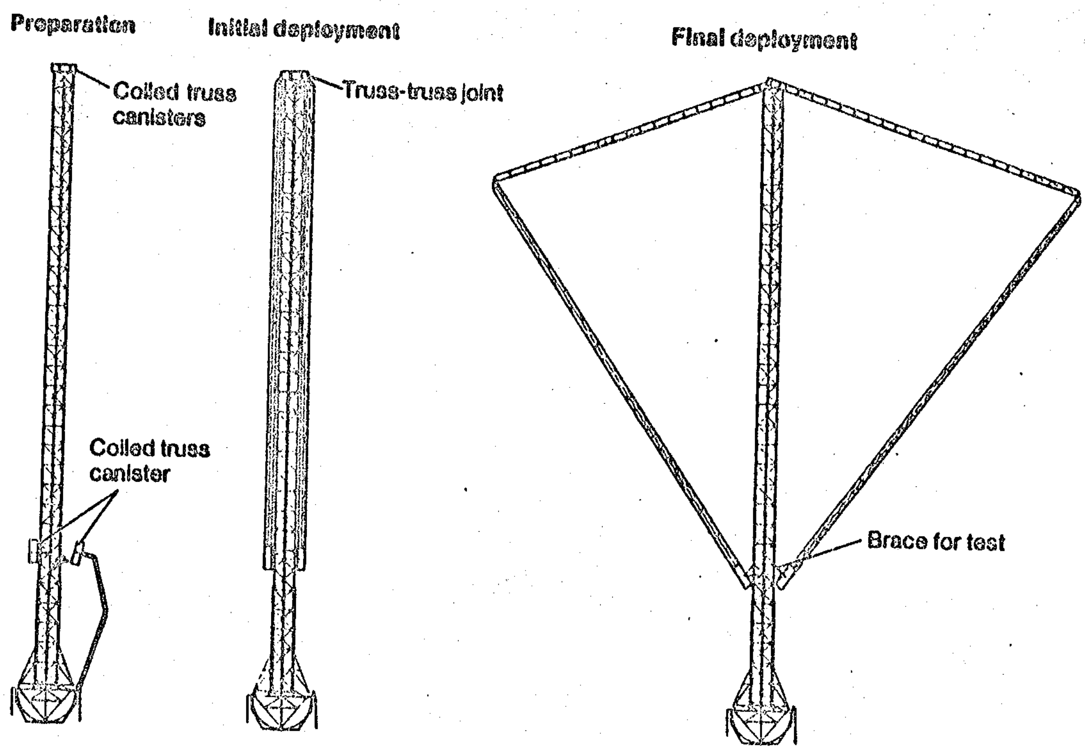


Figure 7-7. Optional Construction Experiment Concept 2

Although this option has merit, its principal disadvantages are:

a. Higher cost due to more extensive hardware requirements.

b. May impact stowage space available to the SCE.

c. This type of construction has no specific applications to which it can be related.

7.7 INSTRUMENTATION PLAN

Instrumentation requirements for the SCE are summarized in Table 7-3.

Table 7-3. SCE Instrumentation Requirements

Measurement Description	Type of Sensor	Qty	Location	Remarks
Starboard Roll Brace Load	Load Cell	1	Cradle Interface	Test Measurement
Port Roll Brace Load	Load Cell	1	Cradle Interface	
Starboard Pitch Brace Load	Load Cell	1	Cradle Interface	
Port Pitch Brace Load	Load Cell	1	Cradle Interface	
Truss Longeron Axial Load	Strain Gauge	12	Truss Bay No. 10, 20, & 31	
Truss Diagonal Strut Axial Load	Strain Gauge	24	Truss Bay No. 10, 20, & 31	
Truss Node Fitting Axial Load	Strain Gauge	24	Truss Bay No. 10, 20, & 31	
Starboard Carriage Motor Temperature	Thermocouple	1	Motor Housing	
Port Carriage Motor Temperature	Thermocouple	1	Motor Housing	
Starboard Latch Actuator Temperature	Thermocouple	1	Solenoid Housing	
Port Latch Actuator Temperature	Thermocouple	1	Solenoid Housing	
Damper Motor Temperature	Thermocouple	2	Motor Housing	
Z-Axis Truss Acceleration	Accelerometer	5	Truss Station 0, 6, 12, 18, 24	
Y-Axis Truss Acceleration	Accelerometer	5	Truss Station 0, 6, 12, 18, 24	
X-Axis Truss Acceleration	Accelerometer	5	Truss Station 0, 6, 12, 18, 24	Test Measurement
Z-Axis Rate	Rate Gyro	2	Damper Z1 & Z2	Test & Control Measurement
Y-Axis Rate	Rate Gyro	2	Damper Y1 & Y2	Test & Control Measurement
X-Axis Rate	Rate Gyro	2	Damper X1 & X2	Test & Control Measurement
Starboard Carriage Position	Encoder	1	Carriage	Control Measurement
Port Carriage Position	Encoder	1	Carriage	Control Measurement
Torque Wheel Response	Encoder	6	Damper Motors	Control Measurement

SECTION 8

PRELIMINARY PROGRAM PLAN

8.1 COST ANALYSIS

Preliminary ROM cost estimates have been developed for a series of candidate space construction experiment concepts, to aid in their evaluation and selection, and a refined estimate has been made for the selected configuration. Because of the limitations on experiment definition and time for the estimating task itself, a parametric modeling approach was used that is well suited to providing timely estimates early in the design definition process.

8.1.1 METHODOLOGY. The parametric cost model used for this analysis is an adaptation of our Space System Life Cycle Cost (SSLCC) model tailored specifically for the Space Construction Experiment (SCE). The SSLCC model was developed in-house over the last several years and used extensively for the SCAFEDS, Geostationary Platform Study, OTV study, and other studies of similar flight vehicles.

Initially a cost-related work breakdown structure (WBS) was developed that includes all elements chargeable to the space construction experiment project for each program phase; specifically development, production, and operations (Section 8.1.3). (Operations costs were not, however, addressed in this study.) This cost WBS then sets the format for the estimating model, the individual cost estimating relationships (CERs), cost factors, or specific point estimate requirements, and finally the cost estimate output itself. Estimates are then made for each cost element either at the breakdown level shown or, in certain cases, one level lower. These estimates are then accumulated to provide the cost for each program phase.

The estimating methodology varies with the cost element and with the historical data or supplier estimate availability, etc. For new non-off-the-shelf hardware, parametric CERs are used. These CERs have been derived for various families of hardware and many subcategories representing differing levels of complexity. They are derived from available historical cost data or detailed estimating information and relate cost to a specific driving parameter such as weight, area, power output, etc. For example, the various experiment structural items were estimated using CERs.

Engineering point estimates were used for specific pieces of known equipment where the definition data were sufficiently detailed or the hardware item was existing equipment and cost data were available. A typical example of this type of estimate is for the ROM estimates for some of the dynamic test equipment (gyros, etc.).

The remaining "floating item" cost elements, such as system engineering and integration, program management, etc., are estimated using cost factors consisting of appropriate percentages of the applicable related program effort.

The nonrecurring or development phase includes all of the one-time tasks and hardware to design and test the experiment. It includes the design and analysis for all ground and flight hardware including structural analysis, stress, dynamics, thermal, mass properties, etc. The nonrecurring category also includes all component development and test through component qualification as well as all component development test hardware. In addition, this phase includes: software development, system engineering and integration; system level test hardware and engineering test prototype and qualification article; and system test. (Since the protoflight approach will be used for this experiment, a single flight vehicle will be manufactured and all system level testing will be accomplished using the flight vehicle, which will then be refurbished and updated to flight configuration.) Also included are GSE design, development, test, and manufacture; facilities; and overall program management and administration.

The production phase (unit cost estimate) includes all tasks and hardware necessary to fabricate one complete set of flight hardware equipment. It includes all material and component procurement, parts fabrication, subassembly, and final assembly. In addition, this category includes the required quality control/inspection task, an acceptance test procedure for sell-off to the customer, and program management and administration activities accomplished during the manufacturing phase.

Operating costs and Shuttle user charges were not included in the cost analysis at this time.

8.1.2 GROUND RULES AND ASSUMPTIONS. The general ground rules and assumptions governing the subsequent cost estimates are:

a. Costs are estimated in constant 1981 dollars.

b. Prime contractor fee is not included.

c. Costs are for the design, development, and fabrication of a single, flyable experiment.

d. All system testing required is accomplished using the flight article hardware which is then refurbished for flight.

e. No mission operations or Shuttle user charges are presently included.

f. The cost estimates presented are rough-order-of-magnitude costs for planning purposes only.

8.1.3 WORK BREAKDOWN STRUCTURE (WBS). The WBS is a comprehensive breakdown of all program life cycle elements categorized or sorted into several levels of hardware and task or function-oriented end items. The WBS, thus, serves to identify the cost elements to be included in the cost analysis task. This WBS contains all of the hardware and tasks associated with Phase C/D development and test, fabrication of the flight hardware, and the activities incurred during the test flight itself. It serves as the basic format for cost reporting and programmatic data, and to organize, plan, and manage the subsequent program. The WBS developed for the SCE is shown graphically in Figure 8-1, and each element is briefly defined below.

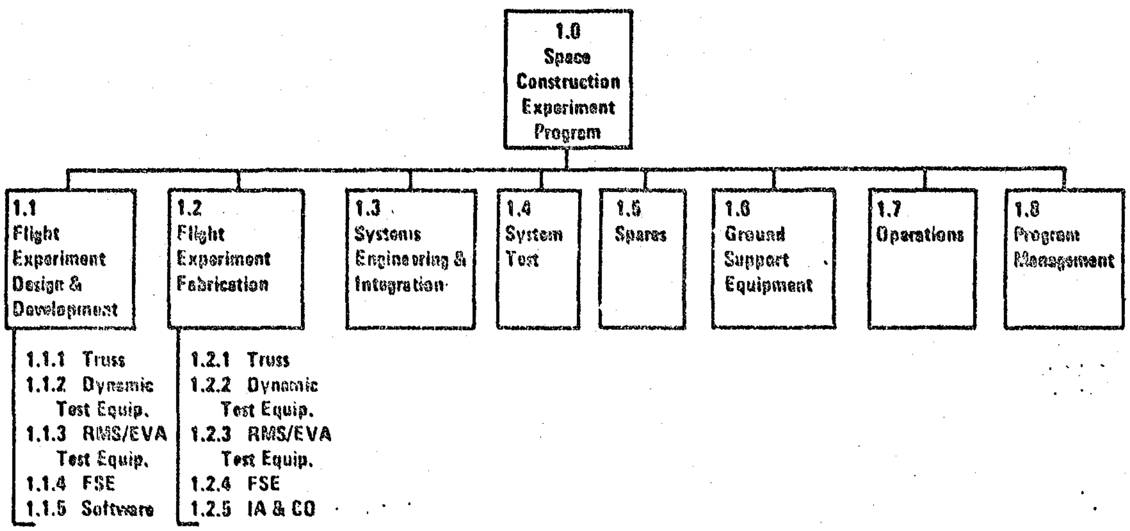


Figure 8-1. Space Construction Experiment WBS

WBS 1.0 Space Construction Experiment Program - This WBS element summarizes all effort and material required for the design, development, fabrication, assembly, test and checkout, and operation of the SCE.

WBS 1.1 Flight Experiment Design and Development - The design and development activities include all tasks and hardware for design and development and testing of the SCE. It includes the required design and analysis for all ground and flight hardware, including structural analysis, stress, dynamics, thermal, mass properties, etc. This nonrecurring category includes tooling, component development, and test through component qualification, as well as all component development test hardware. This element also includes software development.

WBS 1.1.1 Truss - The deployable truss is the primary structural element being tested. It has a diamond cross section and 31 bays and is constructed of composite materials. Also included are the deployment mechanism, experiment support elements, and the tip mass.

WBS 1.1.2 Dynamic Test Equipment - The equipment includes torque wheels and wheel controllers, gyros, accelerometers, microprocessor and logic controller, and their wiring harness. This equipment excites and measures vibrational modes and provides active damping.

WBS 1.1.3 RMS/EVA Test Equipment - The RMS/EVA test equipment includes dummy "black boxes" and attachment fittings, a dummy cabling harness and attach fittings, a portable EVA work stand, and special RMS end pieces.

WBS 1.1.4 Flight Support Equipment (FSE) - The FSE consists of the equipment supporting structure, a data acquisition system and a recorder, and an Orbiter aft flight deck control panel.

WBS 1.1.5 Software - This WBS element consists of all labor, material, and computer resources necessary to provide validated SCE flight software. It includes the design, programming, validation, and verification.

WBS 1.2 Flight Experiment Fabrication - The flight experiment fabrication cost element includes all tasks and hardware necessary to provide one complete set of flight hardware equipment. It includes all material and component procurement, parts fabrication, subassembly, and final assembly. In addition, this category includes the required quality control/inspection task, an acceptance test procedure for sell-off to the customer, and program management and administration activities accomplished during the manufacturing phase.

WBS 1.2.1 through WBS 1.2.4 Subsystems - See Above.

WBS 1.2.5 Integration Assembly and Checkout - This WBS element consists of all effort and materials required to accomplish subsystem installation, final assembly, checkout, and acceptance testing of the platform; and the installation integration, checkout, and acceptance testing of all mission payloads carried on the platform. These are all ground activities and culminate in sell-off to the customer (DD250).

WBS 1.3 Systems Engineering and Integration - This WBS element summarizes all system level studies, analyses, and tradeoffs to support the development of requirements, specification, and interfaces necessary to direct and control the design of the overall system. It also includes all mission studies and analyses to establish requirements and planning for all phases of the mission and logistics activities. It also includes all product assurance activities consisting of safety, reliability, maintainability quality assurance, and parts, material, processes (PMP) control.

WBS 1.4 System Test - This WBS element summarizes all effort and hardware required to conduct and support all major platform and system level testing necessary to refine and validate the design and verify the accomplishment of the development objectives. They may include but not be limited to full scale structural tests, integrated platform avionics tests, all-up platform functional tests, and payload functional and integration testing. This element includes test article refurbishment and reconfiguration; test planning, test analysis, preparation, and test operations; as well as test software and test support activities.

WBS 1.5 Spares - This WBS element includes the procurement and/or fabrication of all spare and repair parts necessary for the development and operational period.

WBS 1.6 Ground Support Equipment (GSE) - This WBS element summarizes all effort and material required to define, design, develop, test and qualify, procure, fabricate, assemble, and check out all GSE required to support the SCE during the development, manufacturing, and operations phase. It includes all necessary handling and transportation equipment, and functional checkout equipment.

WBS 1.7 Operations - This WBS element summarizes all of the effort and materials required to support the SCE project during its operational phase. It includes all ground operation and STS integration activities, flight and mission operations, and operations support.

WBS 1.8 Program Management - This WBS element summarizes all of the effort required to manage, direct, and control the entire SCE program. These functional tasks and activities include planning, organizing, budgeting, scheduling, directing, and controlling other administrative tasks to ensure that the overall objectives of the program are accomplished.

8.1.4 CANDIDATE SCE COST ESTIMATES. During the initial portion of this study, many candidate experiments were examined varying in both size and complexity, driven by differing test objectives. The technical description of these candidate concepts are discussed in Section 4. The principal differences involve mounting position and, hence, support structure, the inclusion of RMS/EVA experiments, one concept having a free flying capability, and one concept involving beam joining. The definition of these configurations was then used to input the cost model described above and produce design and development cost and flight article cost. These costs were generated for the most part parametrically based on preliminary hardware size and performance parameters. Using the model in this manner, even with preliminary input definition data, produces credible relative cost data for use in concept comparison evaluations. The costs were estimated at the hardware assembly or component level and are responsive to definition changes at that level.

The results of those analyses are shown in Table 8-1, and the costs for development and flight article fabrication may be seen for each concept. Two deployable beam truss configurations were estimated for most of the concepts, one being a diamond cross-section, and the other a simpler triangular cross-section. The difference in cost varied between 10 and 20% in savings for the triangular beam.

Table 8-1. Alternative Concept Preliminary Cost Estimate (1981 $M)

	Diamond Beam			Triangular Beam			
Concept	Dev	Unit	Total	Dev	Unit	Total	Remarks
1	5.5	1.2	6.7	5.1	1.0	6.1	With RMS/EVA Experiments
2	4.5	1.4	5.9	3.8	0.8	4.6	No EVA Experiments
2A	6.5	1.6	8.1	—	—	—	With RMS/EVA Experiments
2B	11.2	2.7	13.9	—	—	—	Free Flyer
3	4.8	1.5	6.3	4.2	1.1	5.3	No EVA Experiments
4	—	—	—	3.2	0.8	4.0	Highly Simplified Concept
5	5.5	1.7	7.2	5.0	1.6	6.6	Beam Joining Concept

8.1.5 SELECTED EXPERIMENT COST ESTIMATES. Following the selection of the preferred concept (Version 2A) from the candidates examined in the first phase of the study, additional analysis provided increased design definition detail and refined input parameters used in the cost analysis. Using this updated information, new cost estimates were made for the selected SCE as defined. The results of this analysis are presented in Table 8-2. The total cost for the design, development, fabrication, and test of the SCE is approximately $9M. The experiment flight hardware fabrication accounts for about $1.8M and the remaining $7.2M is required for design and analysis, component development and test, system engineering, the system level test, program, and program management. It should be noted that all system level testing and integration is conducted using the flight experiment equipment that is subsequently refurbished to flight configuration. Also included in this design and development cost is software at $0.2M, ground support equipment at $0.25M, and spare and repair parts at $0.2M.

The majority of the hardware design and development cost is required for structure and mechanisms including the truss itself, its deployment mechanism, and the supporting structure (FSE) for mounting the SCE in the Shuttle payload bay. The dynamic test equipment is considered as virtually all off-the-shelf equipment such as gyros and accelerometers and very little in the way of component development will be required. Only a nominal cost allowance is required for the RMS/EVA test equipment in that there are mass and form mockups only to establish the feasibility of attaching equipment to the truss beam.

Table 8-2. Preliminary ROM Cost Estimates

	COST (1981 M$)		
	Design & Develop		Fabrication
Flight Hardware			
Structure	2.10		0.92
Dynamic Test Equipment	0.48		0.23
RMS/EVA Test Equipment	0.18		0.07
Flight Support Equipment	1.98		0.33
Assy, Integration, & C/O	—		0.09
Software	0.20		—
System Engineering & Integration	0.43		—
System Test	1.00		0.09
GSE	0.25		—
Spares	0.19		—
Facilities	0		—
Program Management	0.34		0.09
TOTAL	7.15		1.82
GRAND TOTAL		8.97	

Operations costs were not estimated at this time but would consist of transportation (to KSC), and ground operations for preparation for STS installation and postflight disposition plus support activities during the flight itself.

8.1.6 ANNUAL FUNDING REQUIREMENTS. Annual funding requirements by fiscal year for development and flight article fabrication were generated by spreading individual cost elements in accordance with the subsequent program schedule discussed below. These annual funding requirements for the selected SCE are presented in Figure 8-2.

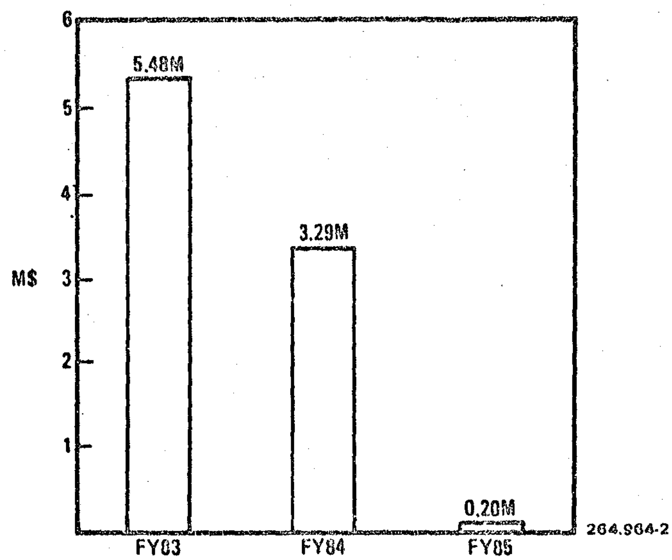


Figure 8-2. SCE Annual Funding Requirements

8.2 PROGRAM DEVELOPMENT PLAN

Based on the overall program scope of this Space Construction Experiment (SCE) and the desired milestones, a summary program development schedule was established. This program master schedule is shown in Figure 8-3.

The approach used to develop the master schedule was to first establish the overall program milestones. All major functional task areas were then identified, together with the necessary sequence of major activities and events. These were to include the sequence of functions and tasks required for each of the principal phases: experiment development and test, flight article fabrication, and the operational flight. Once these major milestones and tasks were identified, detailed program milestones, task durations, and other pertinent data were laid out in the master program schedule. The key activities of each functional task area discipline are identified and time-phased relationships to each other and to the external program milestones such as Shuttle activities are determined. The interfaces and relationships between these activities and the program milestones were thus identified. This program master schedule serves as a focal point for displaying and evaluating interface constraints and time-critical elements.

The overall design and development schedule for this experiment provides for a 24-month development program leading to the flight test in November 1984 as an earliest flight opportunity. Examination of STS traffic model (STS Flight Assignment Baseline - dated 15 December 1980) has identified several opportunities to fly the SCE over the 15 months following November 1984. The schedule is, however, keyed to this initial flight opportunity.

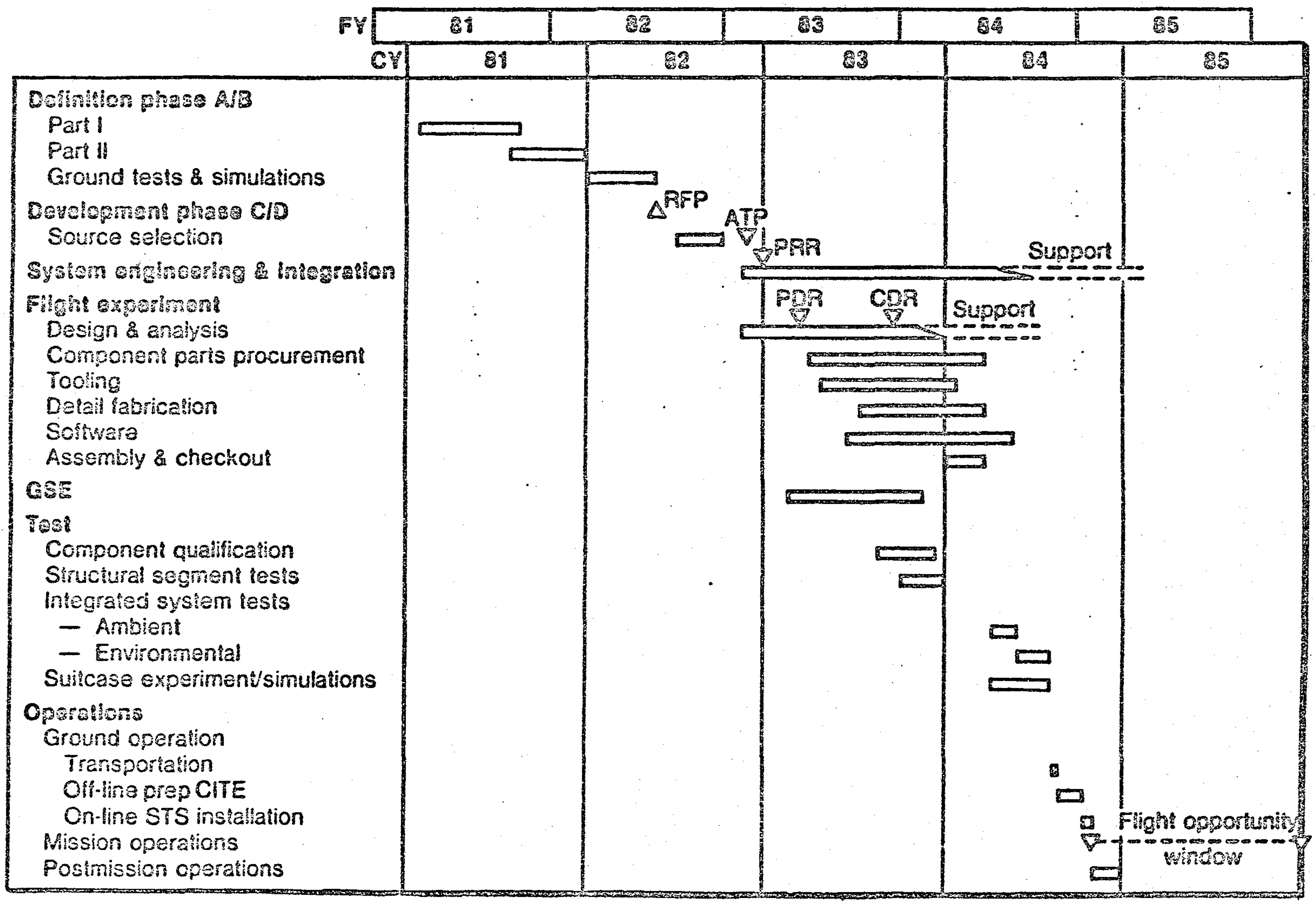


Figure 8-3. Preliminary SCE Program Development Schedule

The 24-month development period is judged to be tight but achievable provided it is preceded by a Phase A/B definition phase in 1981 and certain ground tests and simulations in 1982.

This Phase A/B information will provide refinement of selected concept and tradeoffs, system design data including preliminary systems specifications, and a set of implementation plans including manufacturing, procurement, test, and reliability and safety areas. In addition, schedule and resource estimates will be produced. The principal outputs from these Phase A/B activities are validated requirements, a design solution and supporting analyses, program plans, and a preliminary estimate of resource requirements. The ground tests and simulations envisioned include RMS simulations and neutral buoyancy tests using the current truss hardware. This information will then provide a firm foundation for efficiently proceeding with the subsequent operational system C/D phase of activities.

The planned SCE development is initiated in late CY 1982. Initial design and analysis and development milestones include a Preliminary Requirements Review (PRR) at six weeks and a Preliminary Design Review (PDR) at three months. The Critical Design Review (CDR) follows PDR by six months. The first tooling is available for the parts fabrication in six months, and the overall experiment fabrication is completed at 16 months. Development and system testing using the flight experiment hardware is completed in about 21 months, at mid CY 1984. System testing of the SCE is preceded by the normal component and assembly testing in support of the development effort as well as the required qualification certifications. The test plan activities associated with the SCE development were previously discussed in Section 7 of this report.

The SCE is then transported to John F. Kennedy Space Center (KSC) for a two-month period for integration processing and installation into the Space Transportation System (STS). This period may be shorter and some of the preparation may be conducted at the contractor plant because of NASA desire to minimize STS cargo on-site residency time at KSC. This period is followed by the operational launch, deployment, and test. After return to earth a nominal postflight time allowance is scheduled to handle disposition of flight experiment and ground support equipment, and to analyze and evaluate the flight test data.

SECTION 9

CONCLUSIONS AND RECOMMENDATIONS

This section presents the major conclusions drawn from the SCEDS Part I study effort and provides recommendations for subsequent program efforts to implement the development plan.

9.1 CONCLUSIONS

9.1.1 STRUCTURE

a. The Convair-developed tetrahedral truss with the diamond cross-section has the broadest range of applicability to future large space systems construction, and is the most representative space structure of the candidates considered.

b. Variations of the tetrahedral truss can provide a triangular or a square cross-section for special purpose applications.

9.1.2 CONSTRUCTION OPERATIONS

a. Maximum use of the RMS to support deployment operations can greatly reduce the cost and complexity of deployable systems.

b. Controlled linear deployment of space structures is a major safety consideration. It also facilitates progressive assembly techniques for LSS elements. For the SCE it allows the control limits of the DAP to be approached slowly with variable structure characteristics.

c. Space testing of a single deployable primary structural truss beam is an essential first step to understanding and predicting the performance and behavior of large space structures attached to the Orbiter because of the cost, complexity, and uncertainties of performing full-scale ground tests.

d. Retraction capability for the SCE will provide a high degree of flexibility in selecting and performing experiment options and provide a reusable flight test capability for future subsystems and construction aids.

e. The SCE will contribute a greater understanding of the effects of structural rattle and backlash.

9.1.3 PRELIMINARY DESIGN

a. The configuration and length of the SCE are greatly dependent upon the primary mission payloads and payload arrangements.

b. More up-to-date mission assignment data are required to confirm the basic experiment design and capabilities.

c. Jettison of the fully deployed experiment may pose a handling problem for the RMS.

9.1.4 ANALYSIS

a. A near-zero CTE is achievable for the SCE structure using graphite/epoxy fittings and tubes.

b. The SCE structure can be designed for worst-case contingency loads at a penalty to cost, weight, and packaging efficiency.

9.1.5 FLIGHT CONTROL ANALYSIS

a. A rigidly mounted 50m long truss with a 400 kg tip mass does not pose a problem for the DAP. Flexible mounting of the structure will allow the DAP to be challenged by the experiment, help reduce loads in the structure, and allow frequencies of the attached structure to be readily adjusted if required.

b. The state estimator is the key item in understanding interactions between a large space structure and the DAP.

c. Use of torque wheel/rate gyro type dampers at the tip of the structure is the most effective way to provide variable damping and structural excitation capabilities for the SCE.

9.1.6 TEST PLAN. Experimental time for EVA construction operations is severely limited by a one-day work plan. Additional EVAs may have to be considered if more operations experiments are to be included.

9.1.7 PROGRAM PLAN

a. A late 1984 flight of the SCE is achievable if a program start is initiated in early 1983 and a compatible mission is available.

b. The total SCE program cost is within the $10M maximum guideline.

9.2 RECOMMENDATIONS

9.2.1 SYSTEMS DESIGN AND ANALYSIS

a. Conduct further evaluation of suitable Shuttle missions for accommodation of the SCE and select the best candidate flights available.

b. Obtain a preliminary flight assignment for SCE.

c. Further develop and define the SCE preliminary design for Shuttle integration.

9.2.2. FLIGHT CONTROL ANALYSIS

a. Analyze a new experiment model with reduced modal frequencies for a range of mounting stiffness.

b. Select an appropriate mounting stiffness and reevaluate truss loads and sizing for prescribed contingency conditions using the Charles Stark Draper Laboratory dynamic simulation.

c. Evaluate a slower state estimater in DAP simulation.

9.2.3 SYSTEM TEST

a. Prepare plans for ground tests and simulations to further develop system requirements.

b. Initiate a ground test and simulation program.

SECTION 10

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